| AD NUMBER |
| --- |
| **ADB010087** |
| LIMITATION CHANGES |
| TO: **Approved for public release; distribution is unlimited.** |
| FROM: **Distribution authorized to U.S. Gov't. agencies only; Test and Evaluation; 06 APR 1976. Other requests shall be referred to Defense Communications Agency, ATTN: Code 800, Washington, DC 20305.** |
| AUTHORITY |
| **DCA ltr dtd 2 Mar 1979** |

AEROSPACE REPORT NO.
TOR-0076(6792)-1, VOL. II

ADB010087

# A Review of Communications Satellites and Related Spacecraft for Factors Influencing Mission Success

## Volume II: Appendices A, B, C

F. W. BUEHL and R. E. HAMMERAND
Group II Directorate
Systems Engineering Operations

17 November 1975



Prepared for
MILITARY SATELLITE COMMUNICATIONS OFFICE
OF THE DEFENSE COMMUNICATIONS AGENCY
Washington, D. C.



Satellite Systems Division
THE AEROSPACE CORPORATION

UNCLASSIFIED

SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

## REPORT DOCUMENTATION PAGE

READ INSTRUCTIONS BEFORE COMPLETING FORM

1. REPORT NUMBER: TOR-0076(6792)-1
2. GOVT ACCESSION NO.:
3. RECIPIENT'S CATALOG NUMBER:
4. TITLE (and Subtitle): A Review of Communications Satellites and Related Spacecraft for Factors Influencing Mission Success, Vol. II, (Appendices A,B,&C)
5. TYPE OF REPORT & PERIOD COVERED: Final
6. PERFORMING ORG. REPORT NUMBER:
7. AUTHOR(s):

F. W. Buehl
R. E. Hammerand

8. CONTRACT OR GRANT NUMBER(s): F04701-75-C-0076
9. PERFORMING ORGANIZATION NAME AND ADDRESS: Aerospace Corp., P. O. Box 92957, Los Angeles, CA 90009
10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS: PE 33127K
11. CONTROLLING OFFICE NAME AND ADDRESS: Defense Communications Agency, Code 800, Washington, D. C. 20305
12. REPORT DATE: 17 November 1975
13. NUMBER OF PAGES: 240
14. MONITORING AGENCY NAME & ADDRESS (if different from Controlling Office): None
15. SECURITY CLASS. (of this report): UNCLASSIFIED
15a. DECLASSIFICATION/DOWNGRADING SCHEDULE:
16. DISTRIBUTION STATEMENT (of this Report)

B. Distribution limited to U.S. government agencies only; covers test and evaluation of commercial and military satellite communications systems. 6 APR 1976 Other requests for this document must be referred to the controlling office (see block 11)

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

A. Approved for public release; distribution unlimited

18. SUPPLEMENTARY NOTES

None

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

Satellites, Reliability, Complexity, Communications Satellites, Spacecraft, Spacecraft Complexity, Spacecraft Lifetime, Satellite Communications.

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

Volume II of a three volume review of communications satellite and related spacecraft programs to identify factors influencing mission success. Includes a summary of military and commercial programs to date evaluating factors which contributed to success or failure of each.

DD FORM 1 JAN 73 1473 EDITION OF 1 NOV 65 IS OBSOLETE

UNCLASSIFIED

SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

SECURITY CLASSIFICATION OF THIS PAGE(When D...

SECURITY CLASSIFICATION OF THIS PAGE(When Data Entered)

Aerospace Report No.
TOR-0076(6792)-1, Vol[redacted] 2

NH

(6)

# A REVIEW OF COMMUNICATIONS SATELLITES AND RELATED SPACECRAFT FOR FACTORS INFLUENCING MISSION SUCCESS

Volume II. Appendices A, B, C

(9) Final rept.


(10) Prepared by

F. W. Buehl ~~and~~ R. E. Hammerand
Group II Directorate
Systems Engineering Operations

(12) 380p.

THE AEROSPACE CORPORATION
El Segundo, California




(11) 17 Nov[redacted]75

Prepared for

MILITARY SATELLITE COMMUNICATIONS OFFICE
OF THE DEFENSE COMMUNICATIONS AGENCY
Washington, D.C.

Contract No. F04701-75-C-0076

(15)



Secondary distribution of this document is not permitted without an appropriate distribution statement approved by the Government Project Officer.

Distribution limited to U.S. Gov't. agencies only; Test and Evaluation; 6 APR 1976. Other requests for this document must be referred to Defense Communications Agency Code 800 WASH, D.C. 20305

405 078

Report No.
TOR-0076(6792)-1, Vol II

# A REVIEW OF COMMUNICATIONS SATELLITES AND RELATED SPACECRAFT FOR FACTORS INFLUENCING MISSION SUCCESS


Prepared

F. W. Buehl
Group II Directorate
Satellite Systems Division

R. E. Hammerand
Group II Directorate
Satellite Systems Division

Approved

H. E. McDonnell, Associate Director
Group II Directorate
Satellite Systems Division
Systems Engineering Operations

B. J. Meyer, Colonel, USAF
Assistant Deputy for Space
Communications Systems


The information in a Technical Operating Report is developed for a particular program and is therefore not necessarily of broader technical applicability.

## FOREWORD

This volume includes supplementary information in support of the discussion and conclusions in Volume I. In particular, Appendix C presents the basic data compiled during the study for analysis of each of the satellite programs. The data are assembled in a form to provide a source of general information on the various satellite programs. The information is organized by program and consists primarily of a program summary, satellite description, program milestones, and on-orbit experience. Not all categories are included for each program since several have not yet had a first launch and the amount of information obtainable on all the programs was not identical. For two programs (Project A and DSCS II), sections on testing are included as examples of satellite testing procedures. References are included where applicable.

The cutoff date for all data was May 1975.

CONTENTS

CONTENTS (Cont.)

## SATELLITE ILLUSTRATIONS

## SATELLITE PROGRAM MILESTONES

# ABBREVIATIONS

| | |
|---|---|
| A | ampere(s) |
| ABM | apogee boost motor |
| AC | alternating current |
| ACM | automatic caging mode |
| ACS | attitude control system (or subsystem) |
| A/D | analog-to-digital |
| ADAC | attitude determination and control (subsystem) |
| AEC | Atomic Energy Commission |
| AFB | Air Force Base |
| AGE | aerospace ground equipment |
| A-hr | ampere-hour(s) |
| AKM | apogee kick motor |
| AM | amplitude modulation |
| ANC | active nutation control |
| APL | Applied Physics Laboratory, Johns Hopkins University |
| APS | auxiliary propulsion system |
| APT | automatic picture transmission |
| APTGS | APT ground station(s) |
| AQL | acceptable quality level |
| ASSY | assembly |
| ATC | automatic thermal controller |
| ATS | Applications Technology Satellite |
| AT&T | American Telephone & Telegraph Co. |
| AVCS | advanced vidicon camera subsystem |
| BAPTA | bearing and power transfer assembly |
| BCD | binary coded decimal |
| BER | bit error rate |
| BOL | beginning of life |
| BOT | beginning of tape |
| B.S.E. | Broadcast Satellite Experiment (GE) |
| BUV | backscatter ultraviolet (spectrometer) |
| CCT | circuit |
| CDA | Command and Data Acquisition (stations) |
| CDR | critical design review |
| CDU | command data unit |
| cmds | commands |
| CMG | control moment gyro |
| CMOS | complementary metal oxide semiconductor |
| COMM., comm. | communications |
| comsat | communications satellite |
| COMSTAR | domestic communications satellite (AT&T/Comsat Corp.) |
| CONUS | continental (contiguous) United States |
| CR&T | command, ranging, and telemetry |
| CS | Japanese medium-capacity communications satellite |
| CST | combined systems test |
| CTA | control timing assembly |
| CYL | cylinder |
| dB | decibel(s) |
| dBm | decibel(s) relative to 1 milliwatt |
| dBW | decibel(s) relative to 1 watt |
| DC, dc | direct current |
| DCAS | Defense Contract Administration Services |
| DCPI | data collection platform interrogation |
| DCPR | data collection platform reports |
| DCS | data collection system |
| DDP | digital data processor |
| DEA | despin electronics assembly |
| DECEL | despin control electronics |
| DET | direct energy transfer |
| DEV | development |
| DMA | despin mechanical assembly |
| DMSP | Defense Meteorological Satellite Program |
| DoD | Department of Defense |
| DRO | data readout |

# ABBREVIATIONS (Cont)

| | |
|---|---|
| DSCS II | Defense Satellite Communications System, Phase II |
| DSU | data storage unit |
| DTU | digital telemetry unit |
| DUS | Data Utilization Station(s) |
| E | east (longitude) |
| ECHL | earth coverage high level |
| EDT | Eastern Daylight Time |
| EIA | electrical integration assembly |
| EIRP | effective isotropic radiated power |
| EMC | electromagnetic compatibility |
| ENGR | engineering |
| EOL | end of life |
| EPS | energetic particle sensor |
| ERP | effective radiated power |
| ERTS | Earth Resources Technology Satellite (now LANDSAT) |
| ESAE | electronic shaft angle recorder |
| ESM | equipment support module |
| ESMR | electrically scanning microwave radiometer |
| ETR | Eastern Test Range |
| ETV | educational television (ATS experiment) |
| EXP | experiment |
| FAB | fabrication |
| FCC | Federal Communications Commission |
| FDM | frequency division multiplex |
| FLT | flight |
| FLTSATCOM | Fleet Satellite Communications System |
| FM | frequency modulation |
| FMEA | failure modes and effects analysis |
| FPR | flat plate radiometer |
| FSK | frequency-shift keyed |
| FT, ft | feet |
| FWS | filter wedge spectrometer |
| g | acceleration of gravity |
| GAPSAT | (same as MARISAT) |
| GE | General Electric Co. |
| GEA | gimbal electronics (drive) assembly |
| GFE | government furnished equipment |
| GHz | gigahertz |
| GMT | Greenwich Mean Time |
| GOES | Geostationary Operational Environmental Satellite |
| GSE/TD | general systems engineering/ technical direction |
| GSFC | Goddard Space Flight Center (NASA) |
| G/T | gain/temperature ratio |
| HDRSS | high data rate storage subsystem |
| hi-rel | high-reliability |
| hr | hour(s) |
| HRIR | high resolution infrared radiometer |
| HT | height |
| IC | integrated circuit |
| IDCS | image dissector camera subsystem |
| IDCSP | Initial Defense Communications Satellite Program |
| IF | intermediate frequency |
| IM | intermodulation |
| IMU | inertial measurement unit |
| IN, in. | inch(es) |
| Intelsat | International Telecommunications Satellite Consortium |
| IR | infrared |
| IRIS | infrared interferometer spectrometer |
| IRLS | interrogation, recording, and location subsystem |
| IST | integrated systems test |
| ITOS | Improved TIROS Operational Satellite |
| ITR | incremental tape recorder |
| JCE | jet control electronics |
| kbps | kilobit(s) per second |

## ABBREVIATIONS (Cont)

| | |
|---|---|
| kHz | kilohertz |
| km | kilometer(s) |
| LB, lb | pound(s) |
| LES | Lincoln Experimental Satellite |
| LMSC | Lockheed Missiles and Space Company |
| L.O. | local oscillator |
| MARISAT | maritime communications satellite |
| MAS | Ministry of Aviation Supply (UK) |
| max | maximum |
| Mbps | megabit(s) per second |
| MDA | mechanically despun antenna motor drive assembly (Skynet II) |
| MeV | million (mega) electron volt(s) |
| MHz | megahertz |
| MIL STD | military standard |
| min | minute(s) |
| MIT | Massachusetts Institute of Technology |
| MO | month(s) |
| MPA | master purchase agreement |
| MRIR | medium resolution infrared radiometer |
| MSFN | Manned Spaceflight Network (NASA) |
| MSS | multispectral scanner |
| MTTF | mean time to failure |
| MUSE | monitor of ultraviolet solar energy |
| mW | milliwatt(s) |
| MWA | momentum wheel assembly |
| N | north (latitude) |
| $N_2$ | nitrogen |
| NASA | National Aeronautics and Space Administration |
| NATO | North Atlantic Treaty Organization |
| NBTR | narrowband tape recorder |
| NE | northeast |
| NEMS | Nimbus E microwave spectrometer |
| NESC | National Environmental Satellite Center |
| $N_2H_4$ | hydrazine |
| NiCd | nickel-cadmium |
| NM, nmi | nautical mile(s) |
| NOAA | National Oceanic and Atmospheric Administration |
| NP | negative-positive |
| NRZ | non-return to zero |
| NW | northwest |
| OPS | operations |
| PA | power amplifier |
| PACU | power amplifier control unit |
| PAM | pulse amplitude modulation |
| PCM | pulse code modulation |
| PCU | power control unit |
| PDA | percent defective allowable |
| PDR | preliminary design review |
| P/L | payload |
| PLACE | position location and aircraft communication experiment |
| PM | phase modulation |
| PMP | precision mounting platform |
| PMT | photomultiplier tube |
| PRI | primary |
| PRN | psuedo-random noise |
| psia | pound(s) per square inch, absolute |
| PSK | phase-shift keyed |
| PWM | pulse width modulation |
| QA | quality assurance |
| QC | quality control |
| QOMAC | quarter orbit magnetic attitude control |
| QUAL, qual | qualification |
| QPSK | quadriphase-shift keyed |
| RBV | return beam vidicon |
| RCE | reaction control equipment |

## ABBREVIATIONS (Cont)

| | |
|---|---|
| RCS | reaction control system |
| RCU | redundancy control unit |
| RCVR | receiver |
| RDU | receiver-demodulator unit |
| RF | radio frequency |
| RFI | radio frequency interference |
| RGA | radiation generator assembly |
| RMP | rate measuring package |
| rms | root-mean-square |
| R/O | readout |
| rpm | revolution(s) per minute |
| RSS | reaction control equipment support structure |
| RTG | radioisotope thermoelectric generator |
| RTTS | real time transmission subsystem |
| RVCF | Remote Vehicle Checkout Facility |
| SA | solar array |
| SAB | Satellite Assembly Building |
| SAMSO | Space and Missile Systems Organization (U.S. Air Force) |
| SATCOM | satellite communications system |
| SCF | Satellite Control Facility (U.S. Air Force) |
| SCR | selective chopper radiometer |
| SCMR | surface composition mapping radiometer |
| SE | southeast |
| SEC | secondary |
| sec | second(s) |
| SEM | scanning electron microscope<br>space environment monitor (SMS) |
| SESP | Space Experiments Support Program (now STP) |
| SGLS | Space-Ground Link Subsystem |
| SHF | super high frequency |
| SIRS | satellite infrared spectrometer |
| SITE | satellite instructional television experiment |
| SLS | switching logic assembly |
| SMS | Synchronous Meteorological Satellite |
| S/N | signal-to-noise ratio |
| SNAP | System for Nuclear Auxiliary Power |
| SPO | system program office |
| SPM | solar proton monitor |
| SR | scanning radiometer |
| SRR | SR recorder |
| STDN | Satellite Tracking and Data Network |
| STP | Space Test Program |
| SW | southwest |
| SYNC | synchronous |
| TACSAT | Tactical Communications Satellite |
| TBU | time base unit |
| T&C | telemetry and command |
| TDA | tunnel diode amplifier |
| TDAL | tunnel diode amplifier/limiter |
| TDM | time division multiplex |
| THIR | temperature, humidity infrared radiometer |
| TI | Texas Instruments, Inc. |
| TIROS | Television and Infrared Operational Satellite |
| TLM, T/M | telemetry |
| TOS | TIROS Operational Satellite |
| TRUST | television relay using small terminals (experiment) |
| TT&C | tracking, telemetry, and command |
| TV | television |
| T/V | thermal/vacuum |
| TWT | traveling wave tube |
| TWTA | traveling wave tube amplifier |
| TYP | typical |
| UHF | ultrahigh frequency |
| UK | United Kingdom |
| USB | unified S-band |

## ABBREVIATIONS (Cont)

| | |
|---|---|
| V | volt(s) |
| VCC | variable command count |
| VCVPD | voltage-controlled variable power divider |
| Vdc | volt(s), direct current |
| VHF | very high frequency |
| VIP | versatile information processor |
| VHRR | very high resolution radiometer |
| VHRRGS | VHRR ground station(s) |
| VISSR | visible infrared spin scan radiometer |
| VISSR DR | VISSR digital multiplexer |
| VREC | visual data recorder |
| VTPR | vertical temperature profile radiometer |
| W | watt(s) |
| WBVTR | wideband video tape recorder |
| WDL | Western Development Laboratories (division of Philco-Ford) |
| WECO | Western Electric Co. |
| WEFAX | weather facsimile |
| WESTAR | Western Union commercial communications satellite |
| W/F | wow/flutter |
| WK | week(s) |
| WT | weight |
| WTR | Western Test Range |
| XMTR | transmitter |
| YR | year(s) |

# APPENDIX A

# SPECIAL PRODUCTION LINE PROGRAMS

## A.1 THE DEFENSE SUPPLY AGENCY'S MILL RUN PROGRAM

The Mill Run Program is a program wherein, without a contract, manufacturers agree to produce, stock, and distribute specific materials. They agree that the materials will meet appropriate federal or military specifications. Production is in anticipation of possible procurement by DoD activities or DoD contractors or subcontractors. Defense Contract Administration Services (DCAS) quality assurance personnel verify compliance with specifications. The material is then marked in accordance with applicable specifications and also with the manufacturers' symbols to insure its identity throughout stocking and distribution. Thus, government source-inspected specification material is readily available, essentially off the shelf, for purchase by DoD procurement offices and by defense contractors.

Those products which are considered appropriate for the program are basic Military Specification or Federal Specification materials or basic parts usually purchased in relatively small quantities. For example, aircraft aluminum and aluminum alloys fall into this category. Recently, MIL-M-38510 and MIL-M-0038510 microcircuits and established reliability resistors have been added. The benefits of the Mill Run Program are considered by the Defense Supply Agency to be:

a. The purchaser can confidently buy products which have been produced, inspected, and tested by the manufacturer in accordance with the detail specifications as verified by government source inspection.

b. Instead of being faced with several months lead time, material is available in stock for immediate shipment to a government buying office or to a defense contractor who needs the specific material to perform on a particular contract.

c. Longer production runs under tight manufacturing and quality controls generally produce parts with higher quality and higher reliability. This is particularly true in the case of microcircuits.

d. Longer production runs generally reduce unit costs.

e. Mill Run participants realize a sales advantage over nonparticipants (see items a through d).

f. The Mill Run Program results in a cooperative venture between participants and the government which is thought to be more efficient than strictly buyer-seller relationships that involve short production runs.

g. DCAS quality assurance efforts are ultimately reduced since large lot production reduces or eliminates piecemeal inspection and verification on many small quantity purchases.

## A.2 NASA/VIKING CONTROLLED LINE

Texas Instruments, Inc. (TI) was selected by the Martin Company (the Viking prime contractor) to produce digital integrated circuits for the Viking program on a controlled line, i.e., with separate facilities and personnel. Major characteristics of the procurement were:

a. Specification Control - All manufacturing specifications from metallization to final shipment were approved by Martin and no changes were allowed without approval from Martin.

b. Operator Selection and Training - No production operator with less than six months experience could be employed on the Viking controlled line. All production operators were certified by TI's quality control organization to be proficient at their assigned responsibility.

c. Traceability - Traceability to the diffusion lot, metallization lot, and assembly lot was maintained on each unit shipped.

d. Material Review Board - Representatives from the TI and Martin quality control organization were assigned to a material review board. There could be no disposition of questionable material without mutual agreement of both parties.

e. SEM Inspection - A sample from each metal evaporation lot was subjected to scanning electron microscope (SEM) inspection for evidence of lifting metal, undercut metal, incorrect oxide steps, and general quality. Martin in-house representatives reviewed results and gave final approval on material.

f. Visual Inspection - Two 100 percent visual inspections were performed on each integrated circuit die by TI manufacturing personnel at 100-power magnification. This was followed by a lot acceptance by quality control personnel to a 0.65 percent acceptable quality level (AQL). After the units were mounted and bonded, pre-seal inspection was performed by TI manufacturing personnel at 40- and 100-power magnification on 100 percent of the units. Quality control personnel then accepted the lot to a 0.65 percent AQL sampling plan. This inspection was followed by a 100 percent inspection by Martin employees at 40- and 100-power magnification. DCAS personnel then sampled the lot at a 0.65 percent AQL. All of the above inspections were performed to the criteria of MIL-STD-883A.*

g. Bonding Control - Each bonding machine was monitored twice per shift. Ten bonding wires from two units were pulled to destruction. One or more bonding wires whose pull strength was less than 2.0 grams would require that machine to be deactivated until the reason for failure was established and corrective actions taken. After sealing, environmental testing, and initial electrical testing, five units from each assembly lot were de-lidded and subjected to bond strength testing. This calculates to 70 bonds per assembly lot that received bond pull testing. These test data were reviewed by Martin quality representatives and any suspicious lots were routed to the material review board for final disposition. All of the above testing was performed by TI quality control personnel.

h. Destructive Physical Analysis - The five units from each assembly lot that were de-lidded and subjected to bond pull testing were also inspected optically at 100-power magnification for evidence of metal corrosion, foreign material, and general workmanship. Any suspicious lot was routed to the material review board.

i. Recorded Electrical Data - All units were subjected to DC electrical testing at +25, +125, and -55°C prior to burn-in. Recorded data were taken on all units at 25°C. After completion of burn-in, all units were retested and recorded data taken at +25, +125, and -55°C. Delta comparisons were then made on the +25°C data. A 2 percent defective allowable (PDA) on the 25°C data was applied to critical parameters.

j. Burn-In - Each unit was subjected to a 240-hour operating burn-in at 125°C.

k. X-Ray - A two-view X-ray inspection was performed on each unit for evidence of poor wire dress, extraneous material, improper die attach, and voiding of die to package interface.

l. Government Bonded Locker - After all processing had been performed and lot history and data had been reviewed by TI and Martin quality representatives, DCAS personnel reviewed the data package and, if acceptable, placed completed units in a government bonded storage area. Units were shipped to various subcontractors after instruction by Martin.

---

*Test Methods and Procedures for Microelectronics (15 November 1974).

m. Backup Material - A predetermined quantity of each integrated circuit die was processed through SEM inspection and stored in wafer form for use in case of unforeseen yield losses during processing. This material was kept in storage until the entire program was completed.

The primary point made by the vendor* as a result of the procurement was that the parts manufacturer must be involved with prime contractors at the earliest possible date if future reliability and cost goals are to be met.

## A.3 THE NASA/GSFC CMOS COMMON BUY AND STOCKING PROGRAMS

The Goddard Space Flight Center (GSFC) Common Buy contract to buy Series CD4000A complementary MOS (CMOS) integrated circuits is a direct outgrowth of the goal to encourage the use of high quality components despite pricing and lead time problems. This goal was achieved because of a management decision at GSFC, at about the same time, to institute an in-house stocking program for high reliability CMOS. Although the stocking program and the contract are now functionally unrelated (except for the fact that the stocking program is one of the buyers under the contract), the two were initially intertwined and neither would have been realized without the other. The stocking program is intended primarily to service in-house designers at Goddard.

The Common Buy contract signed with RCA for the CD4000A series of CMOS integrated circuits permits purchases not only by Goddard but also by any other NASA center, any government agency, or by any of their contractors, provided that any contractor purchases are (only) for direct government contracts. Whereas many users are eligible to procure devices, the contract stipulates that all orders must be placed through and by GSFC, although parts can be drop-shipped directly to any destination. Figure A-1 shows the complete flow for orders, payment, and parts between RCA, GSFC, other NASA centers, other government agencies and contractors.

*Proceedings of the Space System Microelectronics Seminar held 15-16 April 1975 in Los Angeles; to be published.

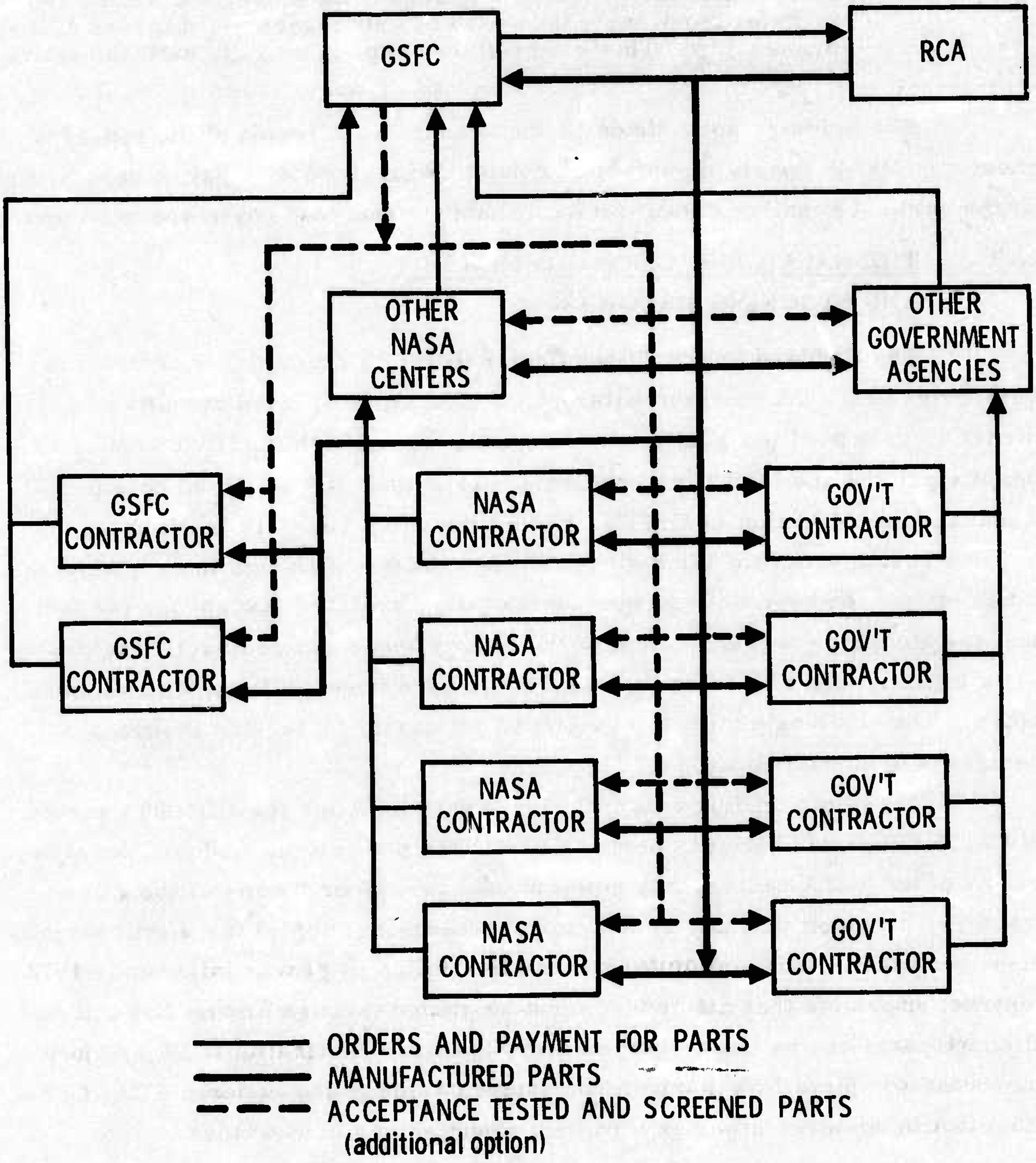



Figure A-1. Flow of Funds and Parts

Under the terms of the contract, GSFC is solely responsible for formally resolving and ruling on problems of any nature (failure mode problems, specification waivers, failure of parts to meet specifications, delivery, etc.) regardless of the delivery destination of an order. Although this status is contractually stated and recognized as a specific responsiblity on the part of GSFC, in actual practice decisions affecting non-GSFC buyers are not autocratic or arbitrary. Other NASA centers, government agencies, or contractors who have placed orders for parts and paid for them through GSFC are expected to accompany the GSFC representative on plant visits. Unless GSFC judges that there are broader implications which, overall, might have a detrimental effect, any decisions made and stated by GSFC on a particular problem will be in compliance with the specific desires of the initiator of that order. However, it must be understood that should GSFC judge that a user's opinions or desires could have repercussions contrary to the purposes of the contract, GSFC can and will rule unilaterally against that user. Uniformity in contract direction and interpretation is logical and necessary, although the implied additional workload on GSFC to resolve technical or administrative problems is undesirable.

Based on the Goddard program it is concluded that common-buy volume contracting is basically a sound approach to obtain high quality piece parts. A stocking program is a natural complement to this buy program and promotes standardization. Standardization is highly desirable because it reduces the piece-parts cost per unit.

## A.4 LMSC MONITORED LINE PARTS PROGRAM

The Lockheed Monitored Line Parts Program was initiated in 1972 for the benefit of Lockheed and its subcontractors. As of April 1975, it has involved six part types and has been applied to 11 programs and 14 major subcontractors. Three part suppliers are participating under a LMSC/ supplier agreement.

The role of LMSC is:

a. Create the Monitored Line Parts Program Office to provide technical and administrative direction.

b. Prepare all specifications and coordinate with users.

c. Maintain the master parts list.

d. Negotiate the Master Purchase Agreement (MPA) with part suppliers.

e. Provide resident monitoring team at each vendor facility.

f. Help establish delivery priorities.

g. Coordinate technical problems with part suppliers and users.

h. Collect subcontractor failure data and analyze for trends and corrective action.

i. Guarantee the sale of part types and quantities.

The role of the vendor is:

a. Supply all parts of a class (e.g., bipolar small-signal transistors) to the requirements of the specification for the total program needs.

b. Allow a team of resident LMSC personnel, who represent Parts Reliability Engineering and Product Assurance Engineering, facility access to perform monitoring functions.

c. Ship electrical test data on each part to its user and to LMSC.

d. Retain all manufacturing traceability records and test data for recall upon demand for a minimum of three years.

e. Provide a Program Manager to coordinate Monitored Line activities and provide a single interface point.

f. Accept orders for designated part types from LMSC and specified contractors.

g. Bill against each using subcontractor order at a negotiated price based upon a block buy.

The role of the subcontractor is:

a. Notify LMSC of part technical requirements so that specifications can be prepared.

b. Provide LMSC with part requirements so that LMSC can assess the total program usage and negotiate guaranteed MPA quantities with the suppliers.

c. Place part orders directly with the part houses.

d. Key all orders to the LMSC MPA.

e. Notify LMSC of part failures at their facility.

Conclusions regarding the program, as presented at a 1975 microelectronics seminar,* were quite favorable. An MPA was found by LMSC to be the best approach to supplying these parts to subcontractors.

*Proceedings of the Space System Microelectronics Seminar held 15-16 April 1975 in Los Angeles; to be published.

## APPENDIX B

## SPACECRAFT BUS CONCEPT

Repetitive development of basically similar subsystems for satellites is both costly and inefficient. The advantages of standardized hardware are obvious and give rise to the concept of a spacecraft "bus" that could be commonly utilized by many space programs. In the following discussion, the advantages, disadvantages, and past precedents for employment of spacecraft buses will be recounted. Additionally, considerations for any bus design for future programs will be discussed.

The spacecraft bus is considered to be a satellite without a payload, but capable of performing a given mission once the payload has been integrated with it. The ramifications of the foregoing statement are that the size and capacity of the bus will be compatible with a range of payloads, and that the performance of bus subsystems can meet all of the unique requirements each of these payloads demands. To satisfy such requirements, it becomes apparent that the bus must have a fairly high degree of flexibility. This flexibility can be achieved in the bus design by adding capacity through modular "building block" design, and with a number of accessory payload-peculiar add-ons to achieve compatibility with the payload requirements.

The advantages and disadvantages of a common spacecraft bus concept are presented in Table B-1. The most obvious advantages are that it avoids the recurring time, cost, and efforts to design and develop basic support subsystems for each project. To some extent this duplication for current space programs is partly avoided by using off-the-shelf components, but the integration of those components into a spacecraft for each particular program is often quite different. Other advantages of standardized subsystems are that time and money are saved by use of standardized procedures for production and testing. Similarly, standardization of subsystem operational interfaces yields savings in procedures for satellite deployment and on-orbit operations. Finally, the accumulation of failure data and corrective measures on the standardized components should improve the quality of the standard subsystems and result in a more reliable spacecraft.

Table B-1. Spacecraft Bus Considerations

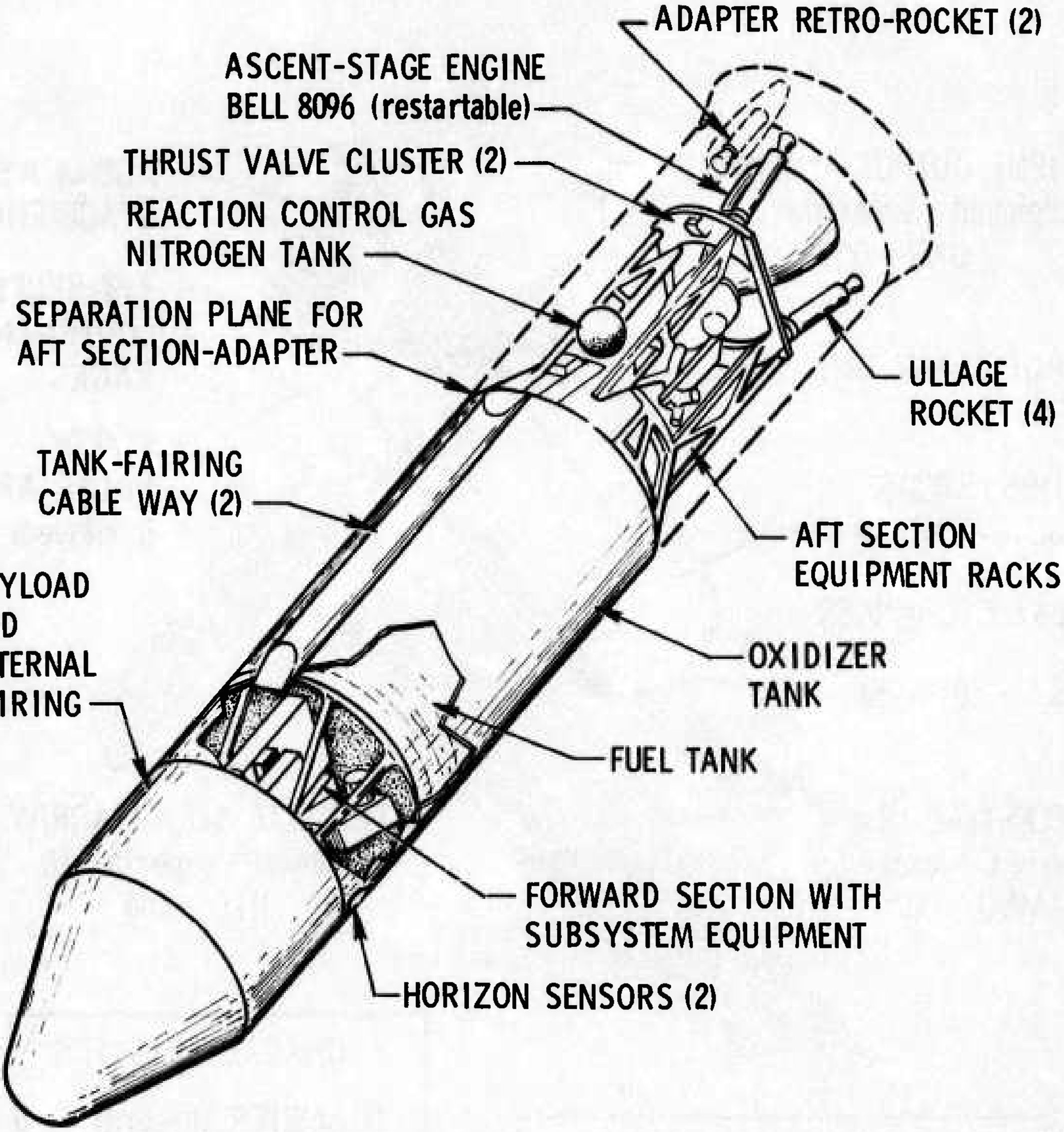
| Advantages | Disadvantages |
|---|---|
| 1. Reduced cost per flight bus<br>2. Reduced time to make production units<br>3. Standard TT&C interface<br>4. Standard interface with launch vehicle<br>5. Improved reliability with experience<br>6. Reduced AGE costs | 1. Bus constraints on payload:<br>a. Size f. Temperature<br>b. Weight g. Attitude<br>c. Power h. External geometry<br>d. Telemetry i. Security<br>e. Command j. Survivability<br>2. High initial cost to amortize<br>3. Reduced overall competitive bidding<br>4. Excess capabilility for some missions |

To realize the above advantages will require a tradeoff in other areas. For example, a given payload will interface to well-defined spacecraft subsystems, but in some cases the payload program may need to accept design constraints imposed by those interfaces, or be willing to pay for additional engineering and test to incorporate an unforeseen requirement of unique nature. In the broader perspective, the very large initial investment to develop the bus, its AGE, and other operational support, may be prohibitive in light of available funding at a given date. Also, once committed to production, subsequent procurement alternatives and competitive bidding are essentially reduced to a sole source. Finally, the bus must always be designed to its worst case mission which means that it is overdesigned for most other missions.

There have been several precedents where a spacecraft bus has been used for more than one program. The most versatile example was the Standard Agena. In 1962, the Air Force Space Systems Division developed a standardized spacecraft with numerous optional equipments to serve as a spacecraft bus. The Standard Agena incorporated a substantial propulsion capability, and could be utilized solely as an ascent stage as well as an orbiting spacecraft. Figure B-1 illustrates the main features of the standard vehicle, and Figure B-2 shows the Agena configured as an orbiting satellite with payloads for the Space Test Program Flight 71-2.

Other Bus concepts similar to Agena were the Burner II upper stage vehicle and the OGO spacecraft.

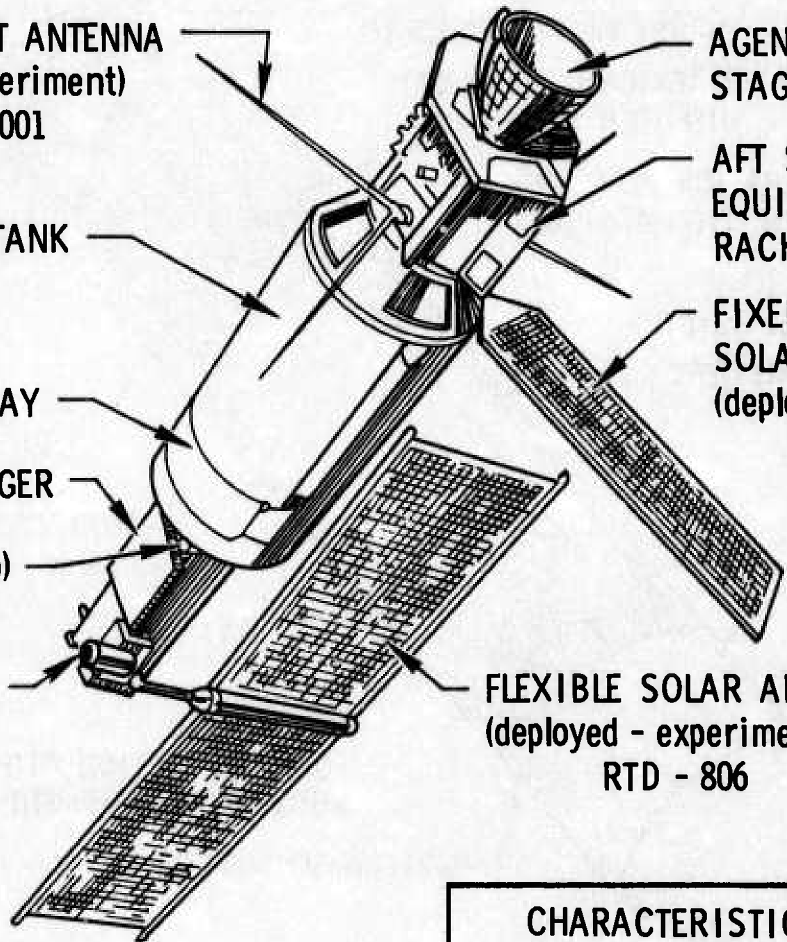
The designs of certain satellites, although not exactly buses, have evolved in a manner that retained many common features of an earlier model. Examples of such derivatives based on a military communications satellite are: Skynet I, NATO II, NATO III, and the commercial Japanese CS. Similarly, there is some design commonality evident in Intelsat IV, Intelsat IVA, and the AT&T COMSTAR. (These configurations are illustrated in Appendix C.) The Intelsat IV also bears a resemblance to TACSAT with the exception of the




| | |
|---|---|
| LAUNCH DATES | - VARIOUS PROGRAMS 1959 THROUGH 1973 |
| BOOSTERS | - ATLAS/AGENA<br>THOR/AGENA |
| ORBIT | - NEAR EARTH (typical) |

| CHARACTERISTICS | |
|---|---|
| DIAMETER | - 60 in. |
| LENGTH (less P/L) | - 243 in. |
| WT (dry-less P/L) | - 1599 lb |
| POWER (batteries) | - 500 W |
| DESIGN LIFE (typ) | 1 wk-6 mo |

Figure B-1. Agena Satellite Vehicle; Standard Agena (SS-01A) Illustrated




| | |
|---|---|
| LAUNCH DATE | - 17 OCT 1971 |
| BOOSTER | - THORAD/AGENA |
| ORBIT | - 425 nmi, 93 deg INCLINATION |

| CHARACTERISTICS | |
|---|---|
| DIAMETER (ascent) | - 5 ft |
| WIDTH (deployed | - 40 ft |
| LENGTH (ascent) | - 31.5 ft |
| WT (on-orbit) | - 3443 lb |
| *POWER (BOL) | - 435W |
| DESIGN LIFE | - 6 mo |

Figure B-2. Space Test Program (STP) Flight 71-2 (Agena Satellite) antennas and AKM. More accurately, the design of Intelsat IV reflects TACSAT experience, in particular the design improvement of the nutation dampers and a change to a brushless drive motor for the despun antennas.

There were several cases where employment of the same spacecraft could be termed a bus. These were the Skynet I/NATO II and the Anik/WESTAR programs. For NASA programs, the use of Nimbus spacecraft design features for the ERTS/LANDSAT satellite represents a near-bus concept.

# APPENDIX C

# SATELLITE PROGRAM DESCRIPTIONS

## C.1 TRANSIT (NAVIGATIONAL SATELLITE)

### C.1.1 Program Summary

The Transit navigational satellite was originally designed and constructed by the Applied Physics Laboratory (APL) of Johns Hopkins University for the U.S. Navy Strategic Systems Project Office. During the period 1962 to 1968, 17 satellites were launched on Scout boosters into 600-mile, circular, near-polar orbits. In 1966, the RCA Astro-Electronics Division at Princeton, New Jersey, became the spacecraft prime contractor. Of 15 flight models built by RCA three have been flown, one each in 1968, 1970, and 1973; 12 are in storage at the RCA plant.

The early Transits had relatively short on-orbit lives. Substantial design modifications (recommended by APL and RCA) to the last three satellites built by APL resulted in much longer life satellites which, in early 1975, were in their eighth year of operation on-orbit. Three of the satellites built by RCA are also still operational on-orbit. One of the RCA-built spacecraft suffered an RF downlink power output reduction that resulted in the satellite being out of operation for a 3-month period and subsequently having a lower than nominal (but satisfactory for operation) system margin. In summary, the system is fully operational with adequate satellites in storage for long term replenishment.

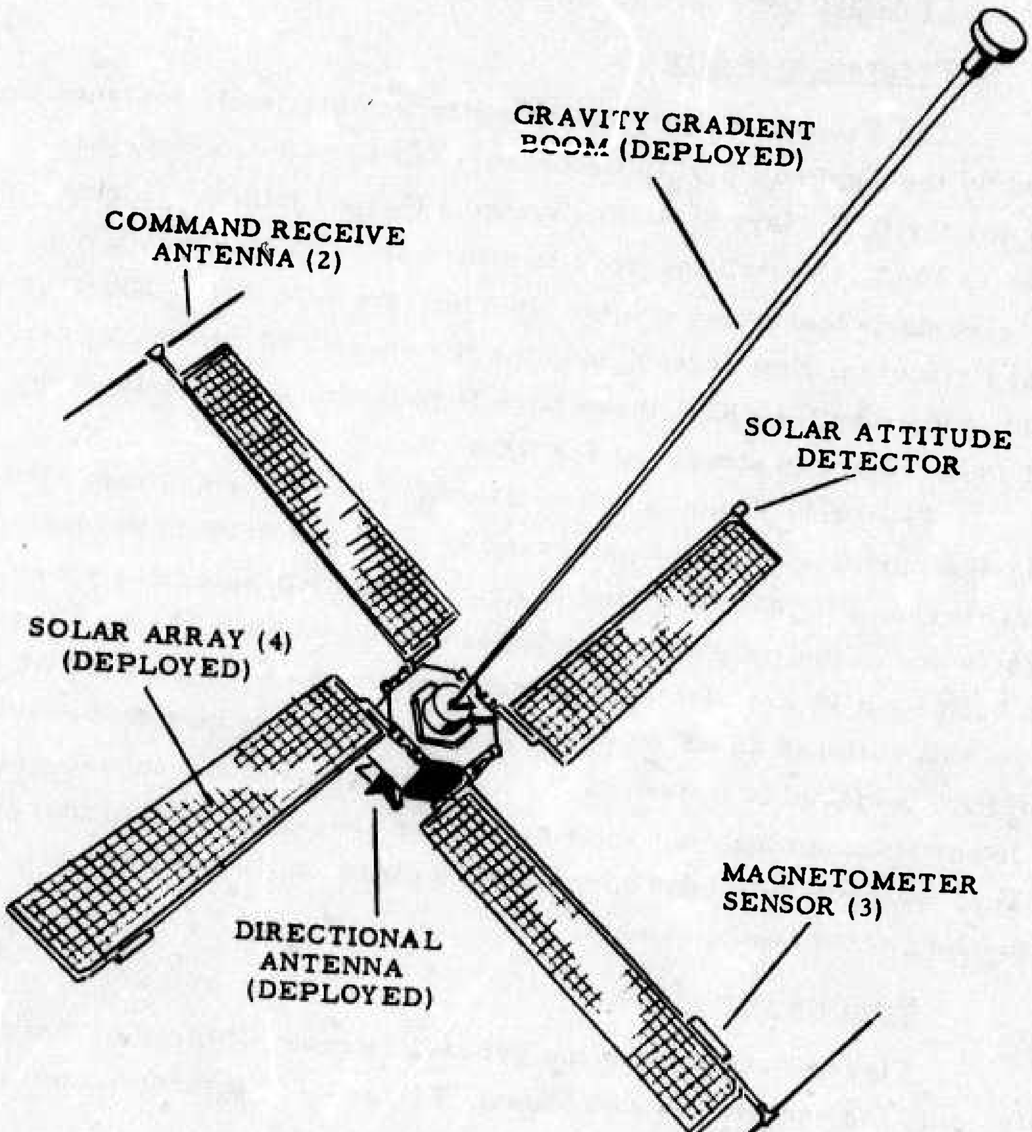
### C.1.2 Satellite Description

Figure C.1-1 shows the general external features of Transit. Internally, only the receiver is redundant. Table C.1-1 summarizes the satellite subsystems.

The spacecraft design concept is one of simplicity. Redundancy is minimal, attitude control is performed by gravity gradient, and deployment mechanisms are uncomplicated. Advanced technology was not employed except in the areas of welded wire, IC memory, and crystal oscillator.

### C.1.3 Key Events and Milestones

Significant program milestones are shown in Figure C.1-2.




LAUNCHES (RCA BUILT)

| | | |
|---|---|---|
| #21 | - | 1 MAR 1968 |
| #22 | - | 27 AUG 1970 |
| #23 | - | 29 OCT 1973 |

BOOSTER - SCOUT
ORBIT - 600 NM POLAR

CHARACTERISTICS

| | | |
|---|---|---|
| WIDTH (ASCENT) | - | 1.5 FT |
| WIDTH (DEPLOYED) | - | 13 FT |
| HEIGHT (DEPLOYED) | - | 102 FT |
| WT (LIFTOFF) | - | 131 LB |
| POWER (BOL) | - | 20 WATTS |
| DESIGN LIFE | - | 1.5 YEARS |

Figure C.1-1. U.S. Navy Navigation Satellite - Transit

Table C.1-1. Transit Subsystem Description

Structure

- Octagon box, 18 in. across
- Innerstructure for strength and equipment mount
- Gravity gradient boom, and sensor attachment structure
- Solar array mounting panels

Orbit and Attitude Control

- Deployed gravity gradient boom (100 ft) and weight
- Hysteresis rods
- Electromagnet

Electrical Power

- Four deployed solar arrays with 13,376 cells
- Four battery modules
- DC-DC converters
- Distribution system

Telemetry and Command

- Two command receivers and antennas
- 35 channels of telemetry at 150 MHz

Thermal Control

- Passive

Payload Subsystem

- Two RF beacons
- Onboard memory for orbital elements
- Directional antenna

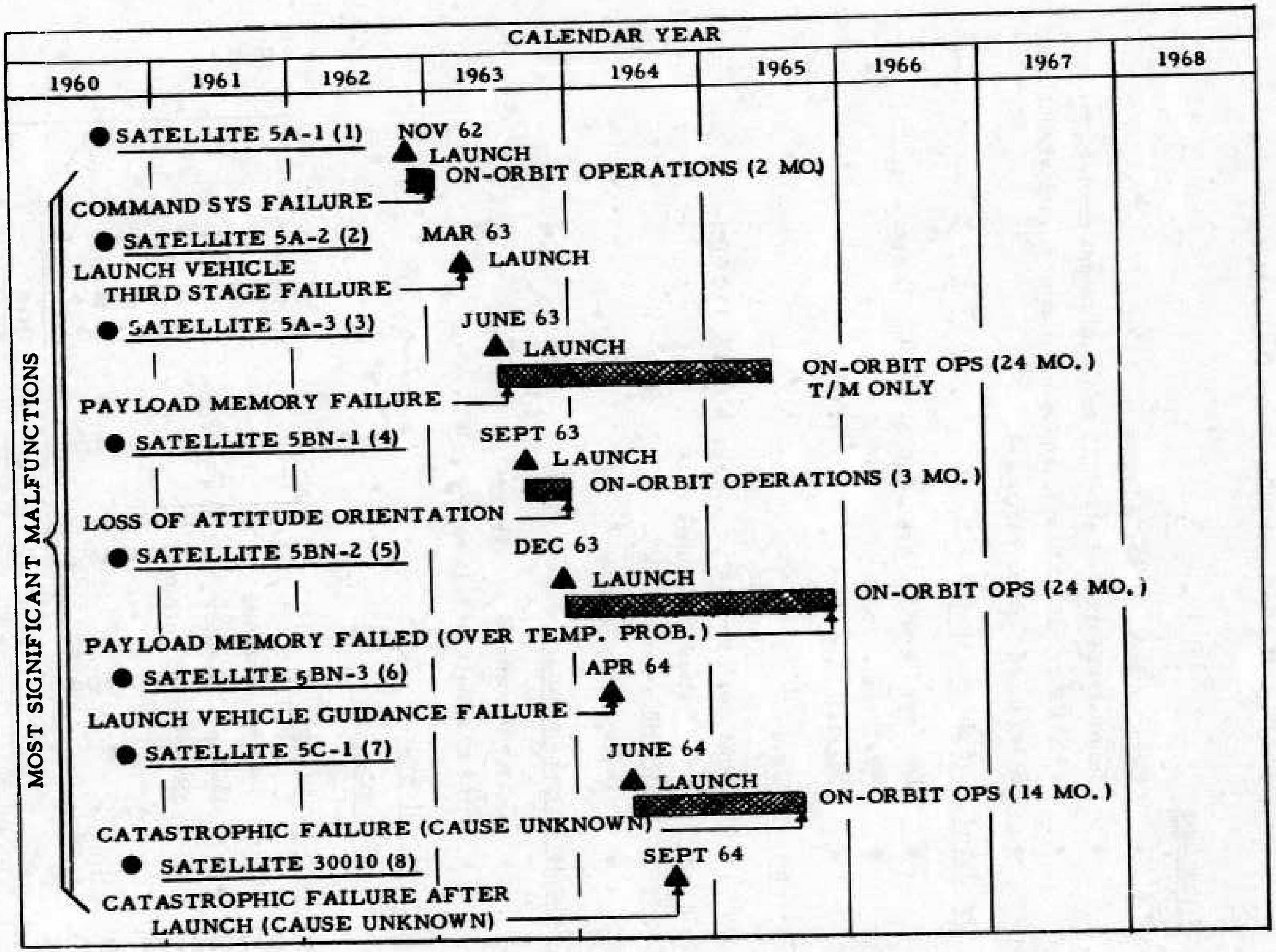



Figure C.1-2. Key Milestones and Events - Transit

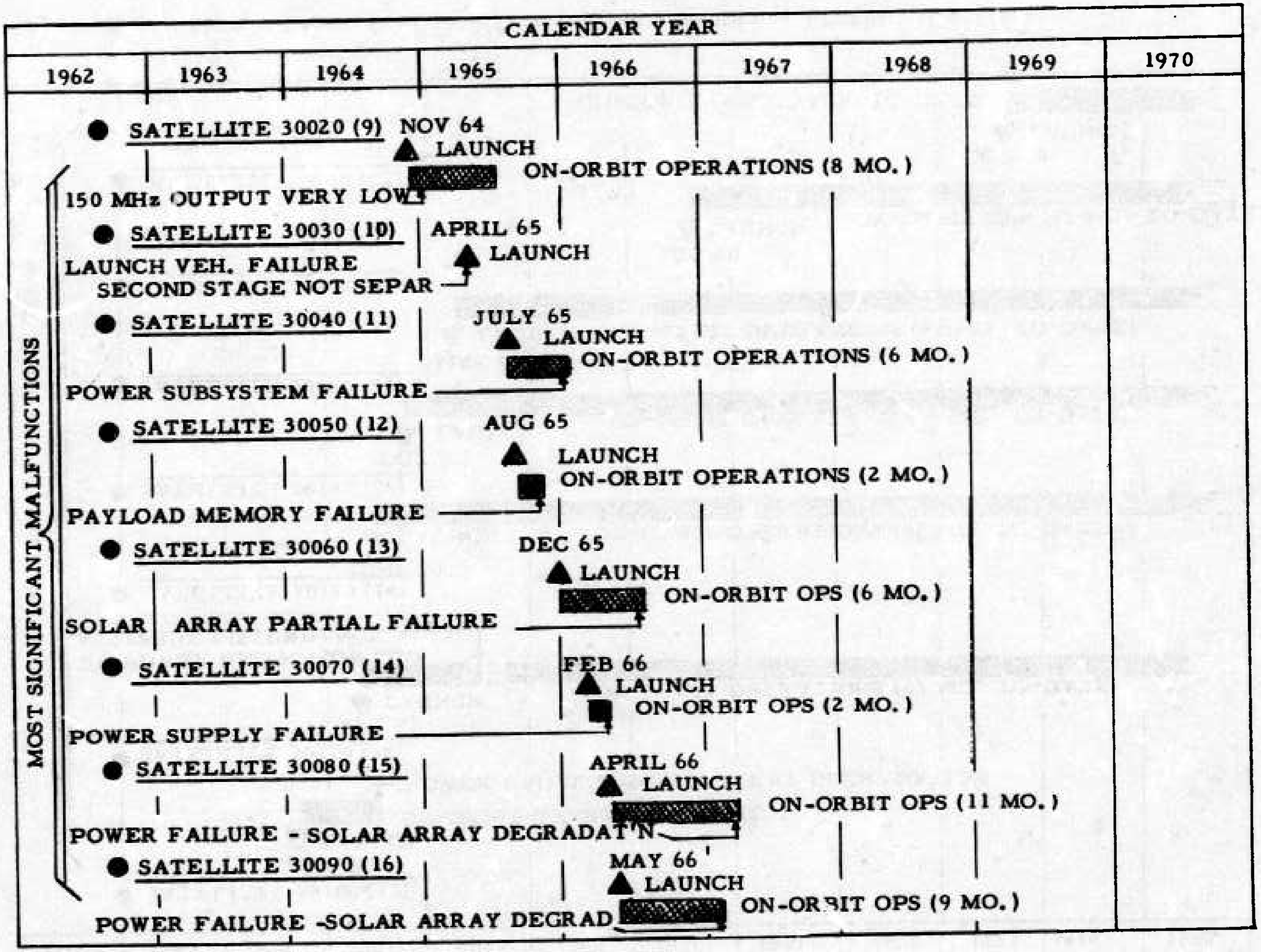



Figure C.1-2. Key Milestones and Events – Transit (Cont.)

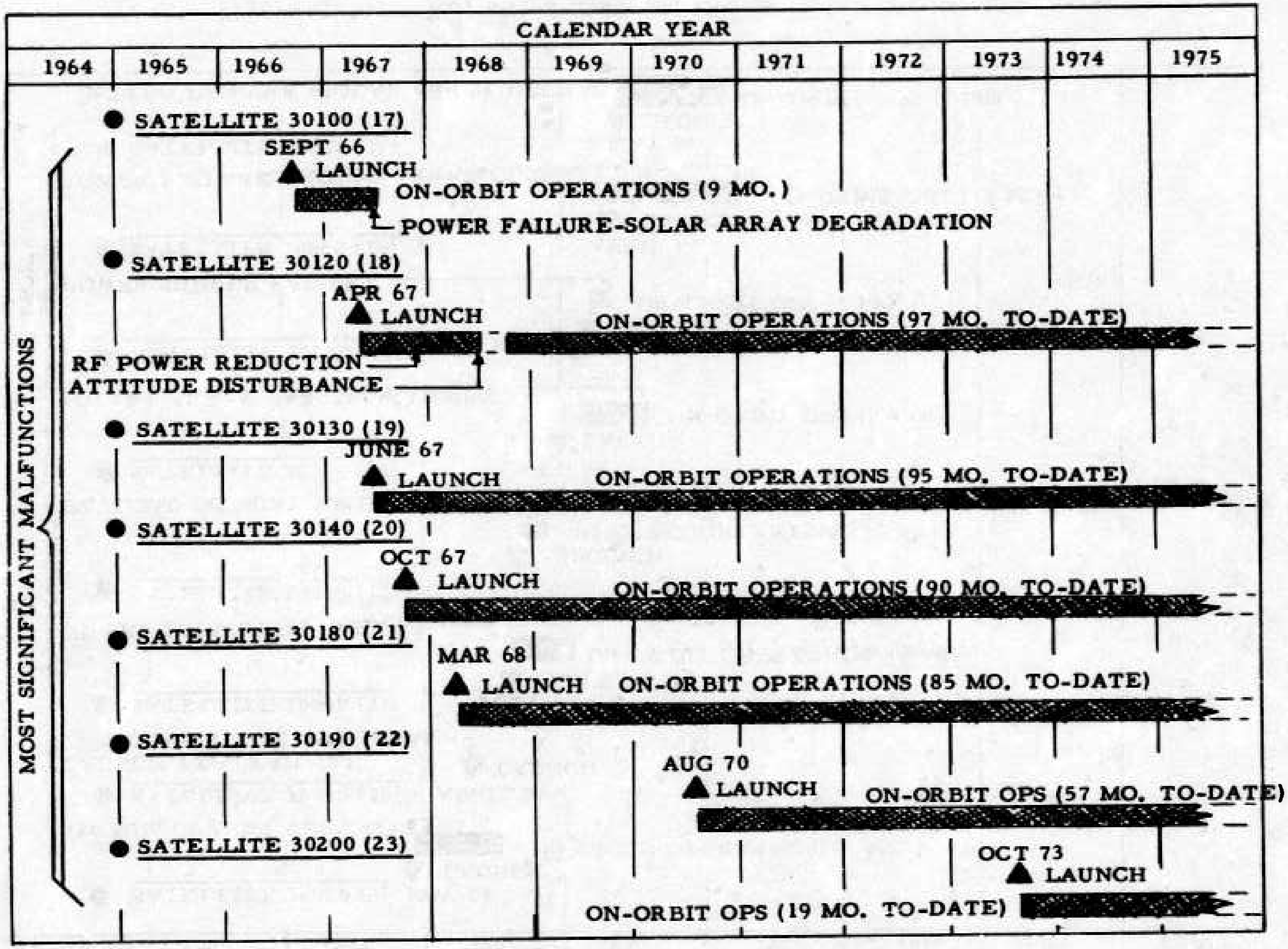



Figure C.1-2. Key Milestones and Events - Transit (Cont.)

C.1.4 On-Orbit Satellite Experience

C.1.4.1 Mechanical

Conservative design in deployment mechanisms resulted in satisfactory operation on-orbit.

C.1.4.2 Electrical

The short life of early models dictated substantial design modifications, which were successful in increasing spacecraft on-orbit longevity. Design simplicity and employment of little advanced technology combined with minimum redundancy was then effective for Transit.

Over-conservatism in part failure rates contributed to much longer than predicted orbital life for the improved vehicles. A secondary effect was the need to store a number of completed spacecraft. No signs have been found of degradation due to storage.

## C.2 AGENA PROGRAMS

### C.2.1 Program Summary

Agena was developed and is fabricated by the Lockheed Missiles and Space Company (LMSC) at Sunnyvale, California. The U.S. Air Force Space and Missile Systems Organization (SAMSO) is the procuring agency.

The first of numerous contracts was started in the late 1950s, with the first launch occurring in February 1959. Several different types of Agenas have been developed, the basic designators being Agena A, S-01 (Agena B), the Standard Agena (Agena D), and Program Assembled Agenas. Agenas have been launched on Thor, Atlas, and Titan boosters. They have been used as booster stages and to carry a variety of payloads. More than 300 Agenas were launched through September 1973. Of these, approximately 200 were Standard Agenas, the first of which was launched in 1962. Program Assembled Agenas have been flown from 1959 to the present time. The discussion herein is primarily related to the Standard Agena and Program Assembled Agenas from Agena flight 256 in January of 1968. Additional data on Agena flights are given in Reference C.2-1 and other documents referenced therein.

### C.2.2 Spacecraft Description

A typical Agena is shown in Figure C.2-1. Items comprising the subsystems are given in the STP 71-2 summary (Section C.13).

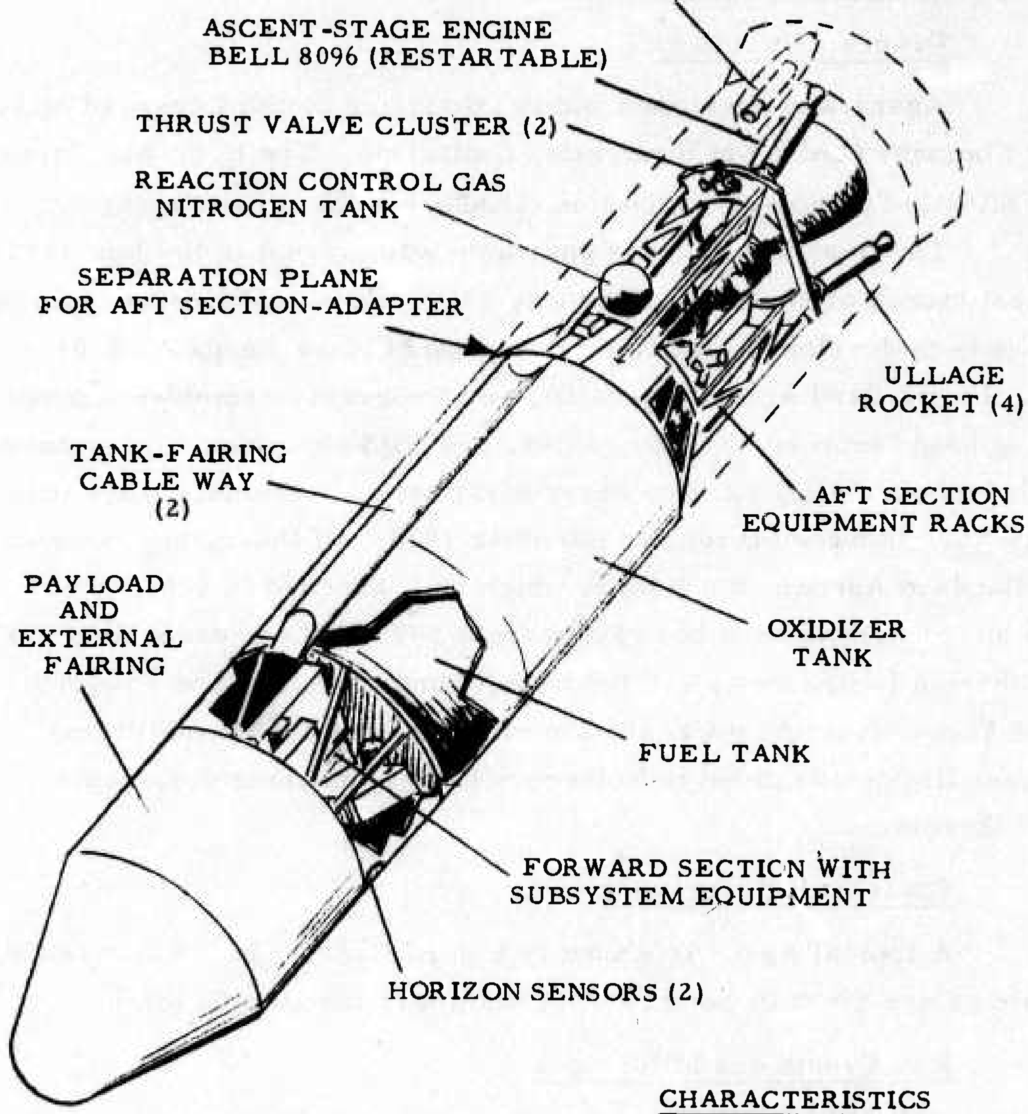
### C.2.3 Key Events and Milestones

Details of the many Agena flights are available from LMSC. A summary of Agena anomalies is given in Reference C.2-2.

### C.2.4 Agena Program Experience

The anomalies experienced by the Standard and Program Assembled Agena models reflect their development from earlier Agenas.

STANDARD AGENA (SS-01A) ILLUSTRATED




LAUNCH DATES -
VARIOUS PROGRAMS
1959 THROUGH 1973

BOOSTERS - ATLAS/AGENA
THOR/AGENA

ORBIT - NEAR EARTH (TYPICAL)

CHARACTERISTICS

| | | |
|---|---|---|
| DIAMETER | - | 60 IN |
| LENGTH (LESS P/L) | - | 243 IN |
| WT (DRY-LESS P/L) | - | 1599 LB |
| POWER (BATTERIES) | - | 500 WATTS |
| DESIGN LIFE (TYP) | | 1 WK - 6 MO |

Figure C.2-1. Agena Satellite Vehicle

The later Agenas had few serious problems that stemmed from the design per se. A large proportion of the anomalies involved part failures and defects traceable to inadequate workmanship. Diodes, valves, commutators, switches, connectors, and relays were the primary type of parts causing the anomalies. Relays were particularly troublesome, with loose particles blamed for many of the anomalies. One solution for this, and other "workmanship" problems, was more stringent testing and inspection. In some cases, an improved type part was procured for subsequent vehicles. On the assembly (component) level, tape recorders were particularly unreliable and numerous changes were made in their design. In some instances changes were made in the vehicle to provide an improved environment for recorder operation.

## C.3 VELA (NUCLEAR DETECTION) SATELLITE

### C.3.1 Program Summary

The Vela satellites were designed and fabricated by the TRW Systems Group at its facility in Redondo Beach, California. This activity was under the direction of the Air Force Space and Missile Systems Organization (SAMSO) and The Aerospace Corporation of Los Angeles.

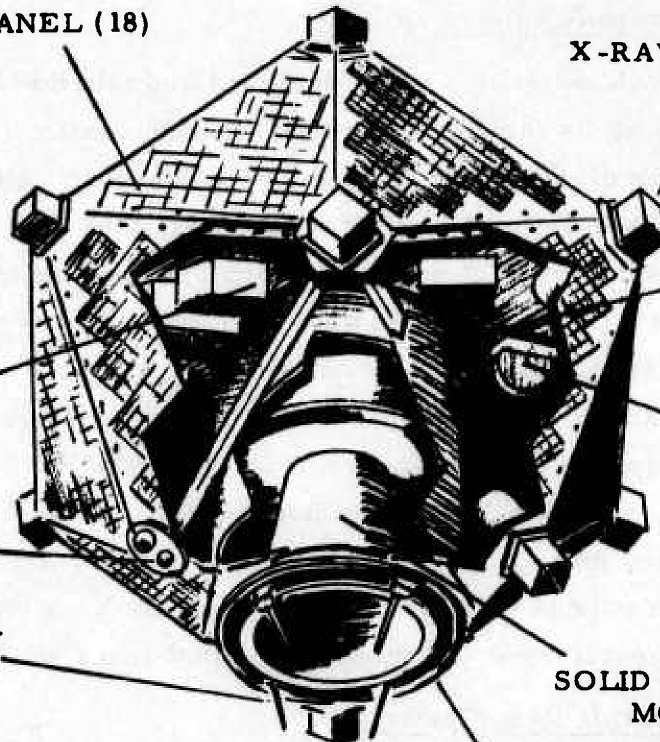
The Vela contract was let in late 1961 and culminated in launch of the first pair of satellites on an Atlas/Agena booster from Cape Kennedy in October 1963. During the course of the program, the design was substantially changed to incorporate improved capability. Subsequent launches on Atlas/Agena boosters occurred in July 1964 and July 1965. Three additional pairs of satellites were launched on Titan IIIC boosters in April 1967, May 1969, and April 1970. All of the satellites enjoyed on-orbit lifetimes far in excess of their design life. Satellites of the first four launches have been deactivated and those of the last two are still operational.

### C.3.2 Spacecraft Description

The general characteristics of the Vela satellite are shown in Figure C.3-1. The icosahedron shape was chosen because it was a good approximation of a sphere, which would best satisfy the Atomic Energy Commission (AEC)* requirement for the location of the external detectors. The 20 equilateral triangular surfaces of the icosahedron were also well suited for relatively simple fabrication operations. The central cylinder of the satellite housed the injection rocket and provided the necessary structural rigidity to enable two identical spacecraft to be launched in tandem. The electronic equipment was supported by the honeycomb-sandwich platform structure, which also provided the heat transfer necessary to maintain the operating temperatures of all components. The various components and their weights are listed in Table C.3-1. A typical weight summary for launching two satellites in tandem on an Atlas/Agena is given in Table C. 3-2.

*Now part of the Energy Research and Development Administration (ERDA).

(VELA I ILLUSTRATED)




LAUNCH DATES -

| | |
|---|---|
| # 1801 1851 | - 16 OCT 1963 |
| # 3662 3674 | - 17 JULY 1964 |
| # 6564 6577 | - 20 JULY 1965 |
| # 6638 6679 | - 28 APR 1967 |
| # 6909 6911 | - 23 MAY 1969 |
| # 7033 7044 | - 8 APR 1970 |

BOOSTERS - FLTS 1, 2, 3 ATLAS/AGENA
FLTS 4, 5, 6 T-IIIC

ORBIT - 60,000 NM
$38^{\circ}$ INCLINATION

CHARACTERISTICS

| | |
|---|---|
| WIDTH - | 58 IN |
| HEIGHT - | 45 IN |
| WT (LIFTOFF)- | 498 LB |
| POWER (BOL) - | 99 WATTS |
| DESIGN LIFE - | 6 MO. |

Figure C.3-1. Vela Nuclear Detection Program Satellite

Table C.3-1. Vela Spacecraft Weight Summary (typical)

| Item | Subitem weight | Weight (lb) |
|---|---|---|
| • AEC Equipment | | 98.5 |
| • Telemetry | | 15.2 |
| • Data storage unit (2) | 6.0 | |
| • Digital telemetry unit (2) | 8.4 | |
| • Signal conditioner | 0.8 | |
| • Communication | | 23.4 |
| • Command decoder (2) | 5.2 | |
| • Diplexer (2) | 3.0 | |
| • Receiver (2) | 8.4 | |
| • Transmitter (2) | 3.3 | |
| • Antennas (2 Ass'ys) | 2.0 | |
| • Coax. cabling and connectors | 1.5 | |
| • Electrical Power | | 69.7 |
| • Battery and supports (2) | 24.5 | |
| • Solar cells, etc. | 25.8 | |
| • Solar panels (structure) (18) | 10.7 | |
| • Converter - AEC (2) | 2.2 | |
| • Converter - transmitter (2) | 2.5 | |
| • Converter - DSU and DTU (2) | 1.3 | |
| • Monitor and control circuits (2) | 2.7 | |
| • Electrical Distribution | | 17.8 |
| • Command distribution unit | 7.9 | |
| • Cabling | 9.9 | |
| • Thermal Control | | 8.2 |
| • Molybdenum heat shield | 0.8 | |
| • Jett. fiberglass heat shield | 4.0 | |
| • Insulation, coatings, etc. | 3.4 | |
| • Structure | | 41.5 |
| • Central structure and attach. | 13.7 | |
| • Equipment platform and supports | 16.3 | |
| • Framework and detector mounts | 11.5 | |
| • Propulsion | | 24.5 |
| • BE-3 engine at burnout | 22.1 | |
| • Engine support structure | 2.4 | |
| • Destruct Subsystem Provision | | 3.0 |

Table C.3-1. Vela Spacecraft Weight Summary (Cont.)

| | | Weight (lb) |
|---|---|---|
| • Balance Weights | | 3.0 |
| • Spacecraft Burnout Weight | | 304.8 |
| • Expendables | | 193.7 |
| • Propellant | 190.7 | |
| • Inerts | 3.0 | |
| Spacecraft Gross Weight | | 498.5 |

Table C.3-2. Dual Vela Payload Weight Summary

| | | Weight (lb) |
|---|---|---|
| • Dual Vela Spacecraft | | 997.0 |
| • Spin Mechanism and Interstage | | 29.8 |
| • Gas | 4.6 | |
| • Tank (1) | 5.7 | |
| • Nozzles, lines, etc. | 3.5 | |
| • Timer, cabling, etc. | 1.3 | |
| • Supports and attach. | 2.5 | |
| • Interstage (1) | 5.2 | |
| • Separation springs | 3.0 | |
| • Spacecraft clamps (2) | 4.0 | |
| • Agena Interstage | | 27.0 |
| • Structure | 17.1 | |
| • Separation springs | 4.4 | |
| • Wiring, hardware, etc. | 2.0 | |
| • Spacecraft clamp (1) | 3.5 | |
| Agena Payload (Less Fairing) | | 1,053.8 |

### C.3.3 Key Events and Milestones

The significant milestones of the Vela program are summarized in Figure C.3-2 along with on-orbit malfunctions.

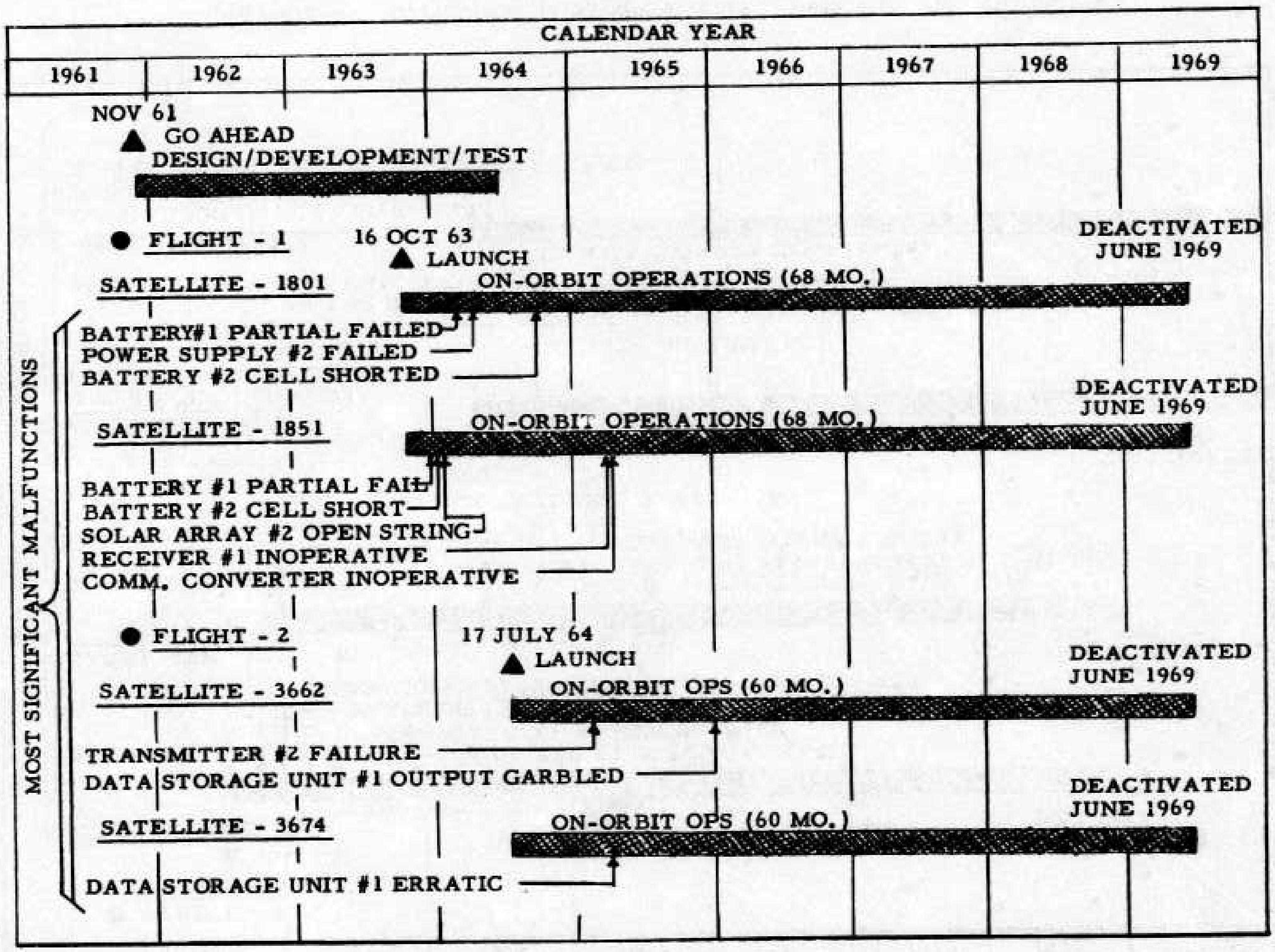



Figure C.3-2. Key Milestones and Events — Vela Program

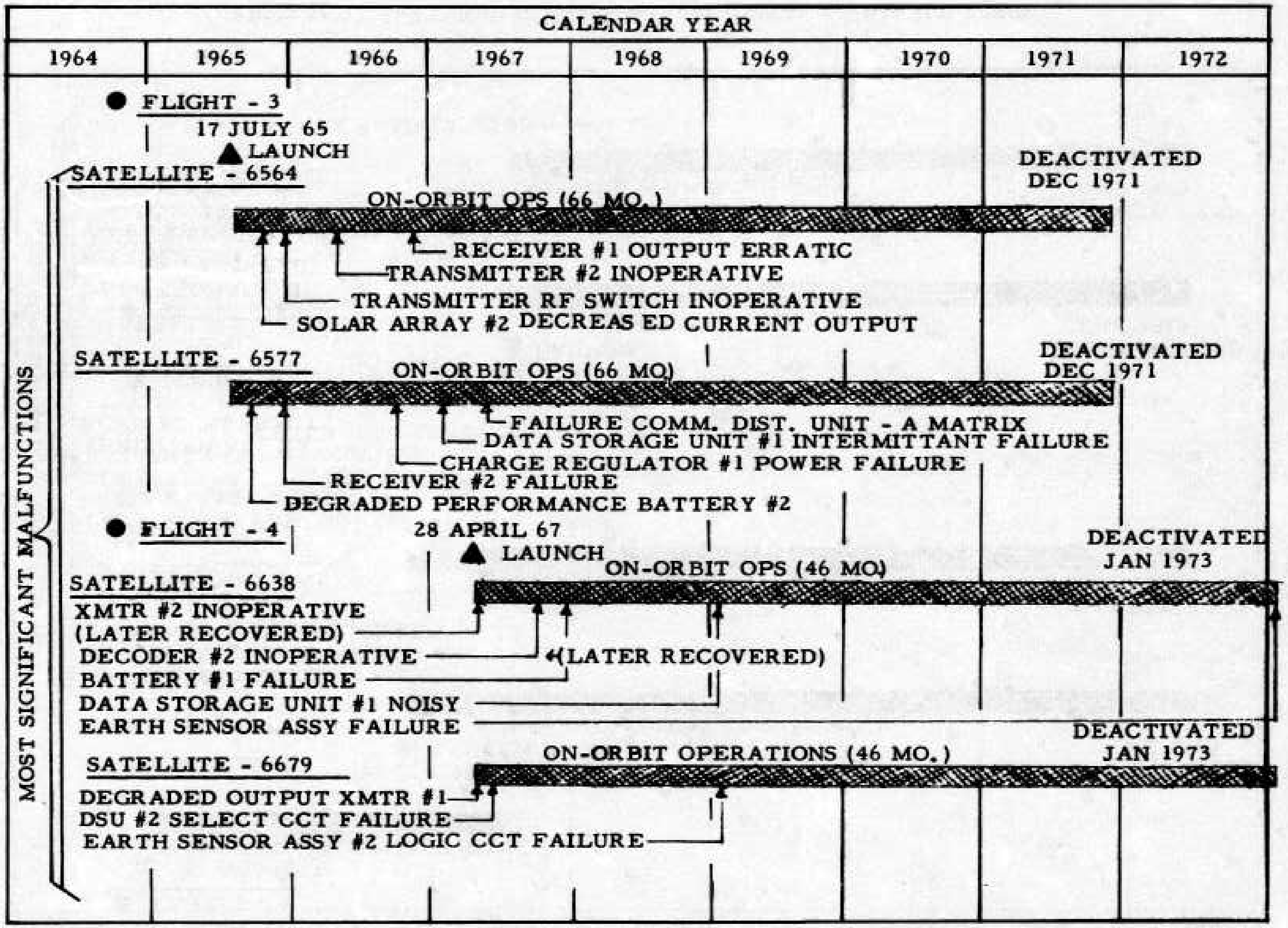



Figure C.3-2. Key Milestones and Events – Vela Program (Cont.)

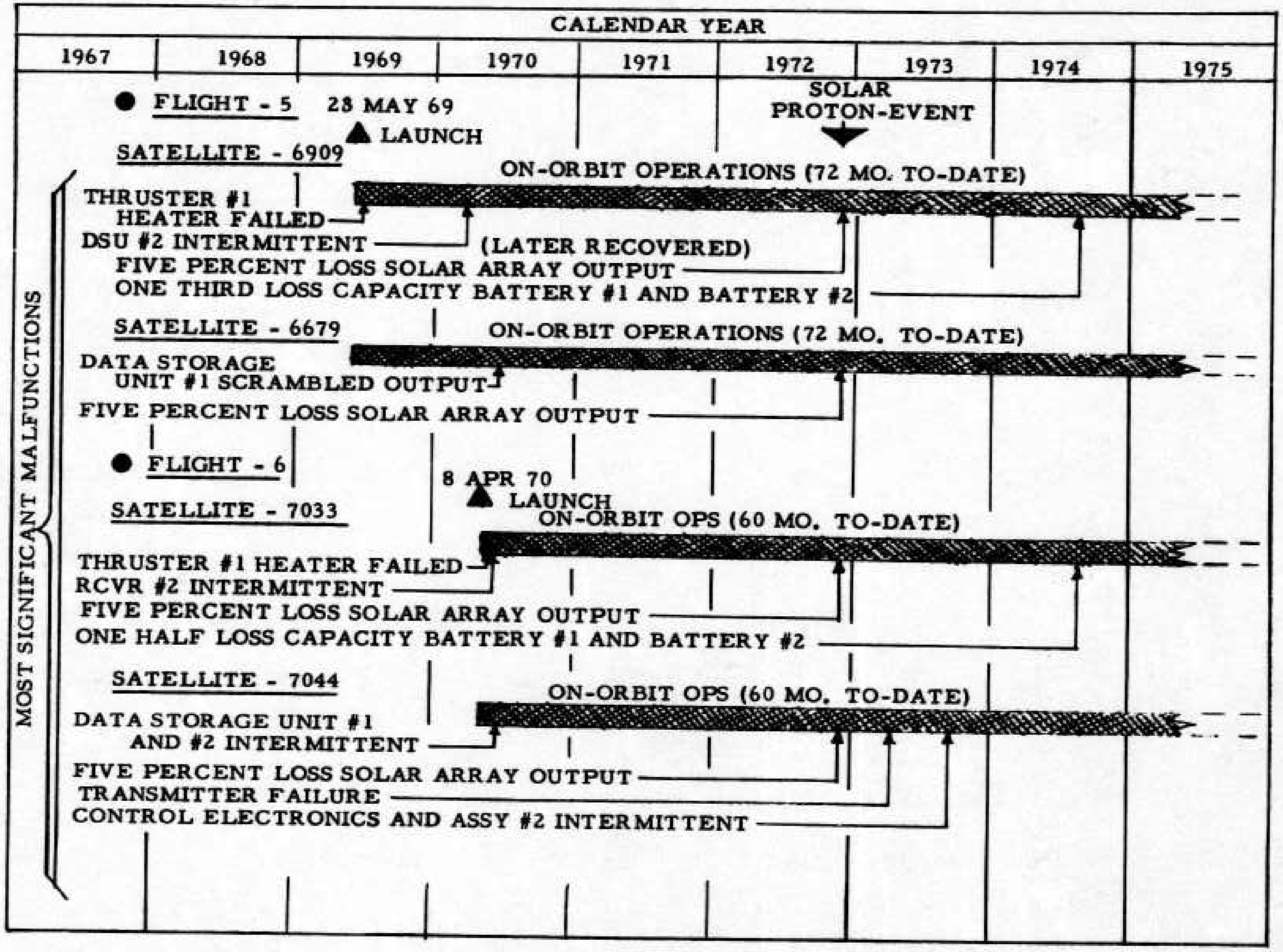



Figure C. 3-2. Key Milestones and Events – Vela Program (Cont.)

## C.4 LINCOLN EXPERIMENTAL SATELLITES (LES)

### C.4.1 General Program Summary

The Lincoln Experimental Satellites are part of a program by MIT Lincoln Laboratory to develop communication satellite techniques, this program having evolved from earlier work by Lincoln Laboratory in ionospheric and tropospheric scatter communications. The basic goals of the LES program include demonstrations of:

a. High efficiency, all solid state transmitters
b. Electronically despun antennas
c. Communications with small mobile terminals
d. Techniques for stationkeeping and attitude control

From the beginning of the LES program through 1972, six LES satellites, all designed and built by Lincoln Laboratory, have been launched. LES 7 was cancelled. The next satellites in the LES series, numbers 8 and 9, are currently under development by Lincoln Laboratory.

### C.4.2 LES 1 and 2

#### C.4.2.1 Program Summary

LES 1 was launched in February 1965. A failure of the Titan IIIA left the satellite in the wrong orbit. Limited tests were run which indicated the repeater and the switched antennas were operating properly. The satellite then entered a tumbling mode which ended its usefulness.

LES 2 was launched on a Titan IIIA in May 1965 and operated as planned for over a year. After nine months in orbit the transmittter power was still 200 mW.

#### C.4.2.2 Satellite Description

LES 1 and 2 were essentially identical. The primary equipment was an all solid state X-band repeater and an eight-horn electronically switched antenna. The transmitter source was a crystal oscillator and multiplier chain which was used for upconversion of the signal from IF; the X-band power was 200 mW.

The eight horns were mounted so as to provide omnidirectional coverage. Sensors were used to determine the direction of the earth and the satellite spin rate. Onboard logic then controlled switches to use the antenna most closely pointed toward the center of the earth. Other details of LES 1 and 2 are given in Table C.4-1.

### C.4.3 LES 3 and 4

#### C.4.3.1 Program Summary

LES 3 and 4 were launched in December 1965 on the same Titan IIIC booster. Due to a launch vehicle malfunction, the satellites were placed in an elliptical synchronous transfer orbit. Initially, the orientation of LES 4 was such that only enough power was available for operation of the telemetry system. Five days after launch the spin axis orientation had changed enough that power was available for all the satellite systems to operate. From that time the LES 4 repeater and antenna operated as expected.

#### C.4.3.2 Satellite Description

LES 3 did not have a repeater; its purpose was to transmit a UHF signal for propagation tests. LES 4 had an X-band repeater of a design similar to LES 2. Improved components significantly lowered the receiver noise figure and increased the transmitter power.

The LES 4 transmitting antenna was composed of eight horns uniformly spaced in a plane normal to the satellite spin axis. Sun and earth sensors and logic circuits controlled the switches to electronically despin the antenna. The difference in antenna design from LES 2 was possible because LES 4 was intended for use in a synchronous equatorial orbit where coverage could be reduced to 26° in the north-south plane. LES 4 technical details are given in Table C.4-2.

### C.4.4 LES 5 and 6

#### C.4.4.1 Program Summary

LES 5 and 6 were built as part of a program to demonstrate the feasibility of using satellites that operate in the military UHF band

Table C.4-1. LES 1 and 2 Technical Details

Satellite

- Polyhedron, about 24 in. each dimension
- 82 lb
- Solar cells, 26 to 33 W initial, no batteries
- Spin stabilized, 180 rpm

Configuration

- 20-MHz-bandwidth triple-conversion repeater

Transmitter

- ~ 7.8 GHz
- All solid state
- 200 mW output, 115 mW at antenna

Receiver

- ~ 8.3 GHz
- 16 dB noise figure

Antenna

- 8 horns, electronically switched (only one used at a time)
- Gain ~ 3 dB

Orbit

- 1500 × 8000 nmi, 32° inclination

Launch Record

- LES 1 - Launched 11 February 1965; launch vehicle failure left satellite in 1500 × 1500 nmi orbit and tumbling
- LES 2 - Launched 6 May 1965

Developed By

- MIT Lincoln Laboratory

Table C.4-2. LES 4 Technical Details

Satellite

- 10-sided cylinder, 31 in. diameter, 25 in. high
- 116 lb
- Solar cells, 36 W initial minimum
- Spin stabilized, 11 rpm

Configuration

- 20-MHz-bandwidth triple-conversion repeater

Transmitter

- ~ 7.8 GHz
- All solid state
- 230 mW at antenna, 3 dBW ERP

Receiver

- ~ 8.3 GHz
- 9 dB noise figure

Antenna

- Transmit: 8 horns electronically switched, 10 dB peak gain, circularly polarized, each horn covered about 26° × 45° of a 26° × 360° toroid
- Receive: biconical horn, 26° × 360°, circularly polarized

Orbit

- Intended: Synchronous equatorial
- Actual: 105 × 18,200 nmi, 26° inclination

Launch Record

- Launched 21 December 1965
- Launch vehicle failure resulted in wrong orbit and orientation
- By 26 December the orientation changed enough to allow sufficient solar cell output for operation.

Developed By

- MIT Lincoln Laboratory

(225 to 400 MHz)* for communications to and from small mobile terminals. Propagation measurements in this band had been made using a previous satellite in the program, LES 3, which functioned as a simple beacon transmitter. LES 5 and LES 6 are similar in many respects. Of the two satellites, LES 6 is the more complex and, from the standpoint of the experimental systems on board, the more interesting.

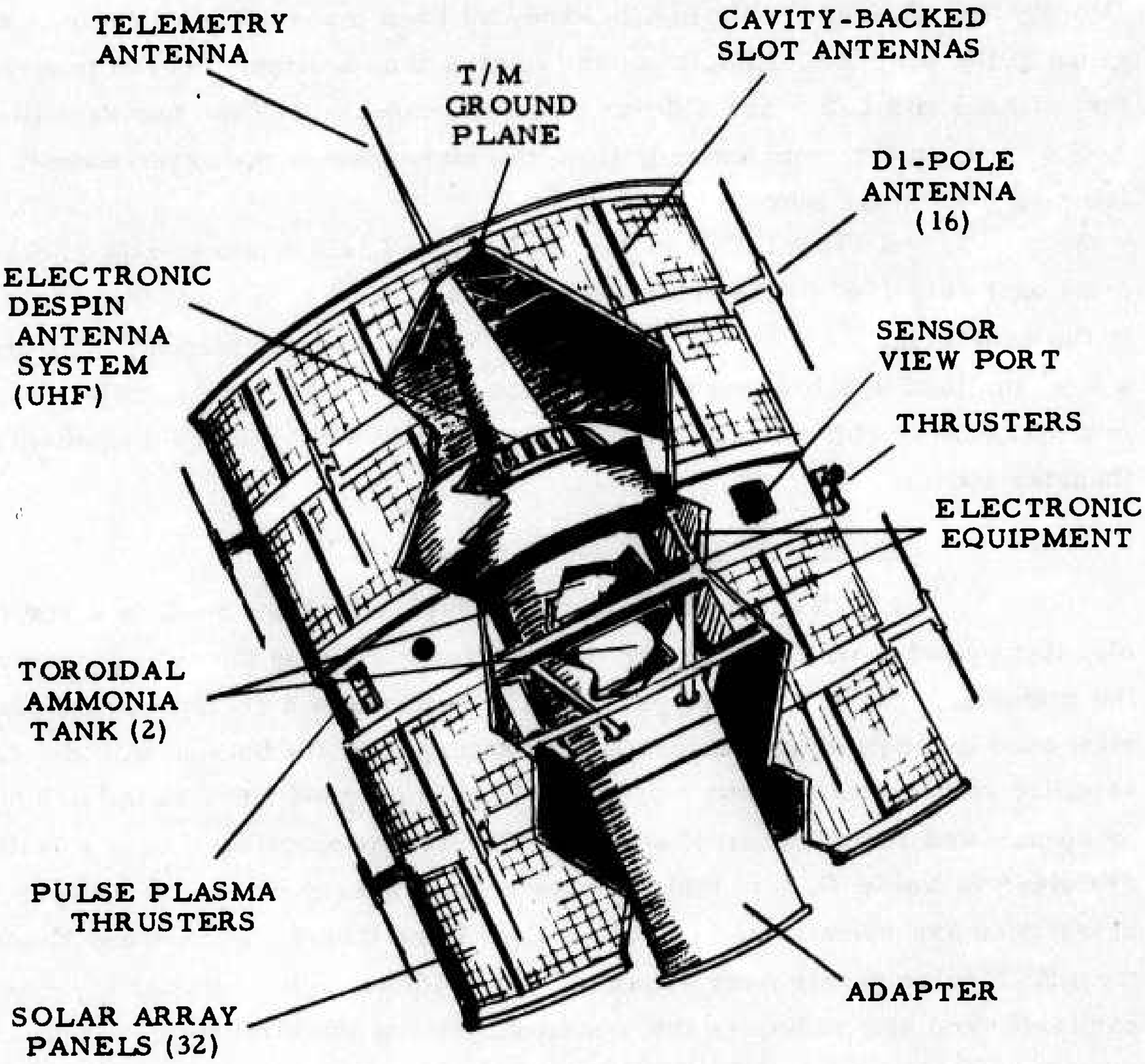
A Titan IIIC launch vehicle placed LES 5 into a sub-synchronous orbit on 1 July 1967. The orbit had a period of 1316.1 min and was inclined to the equatorial plane by 7°. Another Titan IIIC booster placed LES 6 into a 2.9° inclined synchronous orbit on 26 September 1968. Final adjustment into stationary orbit was accomplished by use of an onboard cold ammonia thruster system.

### C.4.4.2 Satellite Description

LES 5 and 6 are depicted in Figure C.4-1. Each is a right circular cylinder with a flat equipment platform dividing the cylinder near the midpoint. Above the periphery of this platform is a section called the view band through which sensors and thrusters view the outside world. Each satellite spins about its long axis which is held perpendicular to the orbit plane by an onboard attitude control system. Physical properties of both satellites are given in Table C.4-3; their subsystems, experiments, and RF characteristics are summarized in Tables C.4-4 and C.4-5. Extensions beyond the LES 5 solar panels were required to provide enough length for a set of cavity-backed slot radiators that constitute half of the circularly polarized antenna system. On LES 6, this area was utilized by solar panels to meet the increased power requirements; the solar array was built in four discrete cylindrical assemblies. The sensor view-band height was also increased from 4 to 6 in. to provide for the large complement of sensors and thrusters. The number of support struts below the deck was increased from six to eight and they were extended to mate with the solar array support fittings to provide

*Called UHF, although the standard designation is VHF from 30 to 300 MHz and UHF from 300 to 3000 MHz.

LES 6 ILLUSTRATED




LAUNCH DATES -

LES 5 - 1 JULY 1967

LES 6 - 26 SEPT 1968

BOOSTER - T-IIIC

ORBIT - SYNC. EQUATORIAL

**CHARACTERISTICS**

DIAMETER - 48 IN

LENGTH - 66 IN

WT (LAUNCH) - 398 LB

POWER (BOL) - 220 WATTS

Figure C.4-1. Lincoln Experimental Satellite (LES 5 and 6)

Table C.4-3. Physical Properties - LES 5 and 6

| | LES 5 | LES 6 |
|---|---|---|
| Diameter, in. | 48 | 48 |
| Length, in. | 64 | 66 |
| Weight, lb | | |
| Prime structure | 22 | 29 |
| Equipment | 115 | 256.5 |
| Solar array and antennas | 63 | 82.5 |
| Dispenser assembly | 22.5 | 25.4 |
| Trim weights | 7 | 4.7 |
| | 229.5 | 398.1 |
| $I_{spin}$, lb in.$^2$ | 72,032 | 113,766 |
| $I_{transverse}$(max) lb in.$^2$ | 58,050 | 96,359 |
| $I_s/I_t$ | 1.24 | 1.18 |
| Spin period, sec | 5.784 | 7.353 |

Table C.4-4. Subsystems and Experiments – LES 5 and 6

| Subsystems | LES 5 | LES 6 |
|---|---|---|
| Telemetry | 100 bps, 8-bit accuracy; 75 channels main frame, 64 channels sub max | 100 bps, 8-bit accuracy; 62 channels main frame, 168 channels sub max |
| Command | 32 cmds, 42 bits each at 0.4 bps | 60 cmds, 47 bits each at 0.4 bps or 6.25 bps |
| Solar array | 30-V series string, 10 Ω-cm N on P cells, 6-mil cover slide, no cell contact protection. 136 W max start of life | 26-V series string, same cells as LES 5, with cell contact protection 220 W max, start of life |
| Antenna despinning | Experimental logic flown, no actual switching | Operational electronic antenna despinning system |
| Stationkeeping | None | Autonomous synchronous orbit E-W stationkeeping ±2° accuracy in longitude |
| Attitude control | Autonomous magnetic stabilization system ±2° accuracy spin axis orientation | LES 5 type mag. stab. system. Autonomous gas thruster system, accuracy variable to ±0.16° |
| Thrusters | None | Cold ammonia gas system for attitude control and stationkeeping; experimental pulsed plasma thruster for stationkeeping |
| Temperature controlled oscillator | None | Dual primary oscillators; temp. control ±0.004°C; frequency drift $< 1 \times 10^{-9}$/day long term |

Table C.4-4. Subsystems and Experiments — LES 5 and 6 (Cont.)

| Experiments | LES 5 | LES 6 |
|---|---|---|
| Solar cell degradation | 5 cells, 2 different types, measure $I_{sc}$ and $V_{oc}$ | 30 cells, 9 different cell types, 5 cover slip types; measure 20-point I-V characteristic each cell vs time, incidence angle; measure cover slip degradation vs time |
| Radiation measurement | None | Advanced LES 4 exp't measure trapped electron spectrum |
| Earth albedo | None | 5 beams, $0.1° \times 0.1°$, scan earth; each measures albedo in 6 spectral bands from 0.41 to 1.00 μm |
| RF interference at UHF | Measure average RF energy and peak-to-average RF energy in 120 kHz steps from 225 to 280 MHz | Same as LES 5 but from 290 to 315 MHz |
| Precision spin period | None | Autonomous spin period measurement |

Table C.4-5. RF Characteristics — LES 5 and 6

| | LES 5 | LES 6 |
|---|---|---|
| Frequency up | 255.1 MHz | 302.7 MHz |
| Frequency down | 228.2 MHz | 249.1 MHz |
| Frequency beacon | 228.43 MHz | 254.14 MHz |
| Transmitter power | 30 W | 120 W[a] |
| Antenna gain (net of filter and matching losses) | 2.5 dB circularly polarized toroidal pattern | 9.5 dB circularly polarized, electronically despun |
| EIRP (measured in orbit) | 45 W | 890 W |
| Receiver noise figure | 3.6 dB | 3.6 dB |
| Receiver IF bandwidth (switchable) | 100/300 kHz | 100/500 kHz |

[a]Varies with maximum available solar bus power. Power amplifiers operate in "optimized" mode, directly from bus.

additional structural rigidity. Some growth capability had been built into the LES 5 structure and this fact, coupled with a nominal reduction of flight loads for LES 6, meant that only minor structural modifications were required to carry the increased payload weight.

### C.4.4.3 Key Milestones and Events

The significant program milestones for LES 5 and 6 are shown in Figure C.4-2.

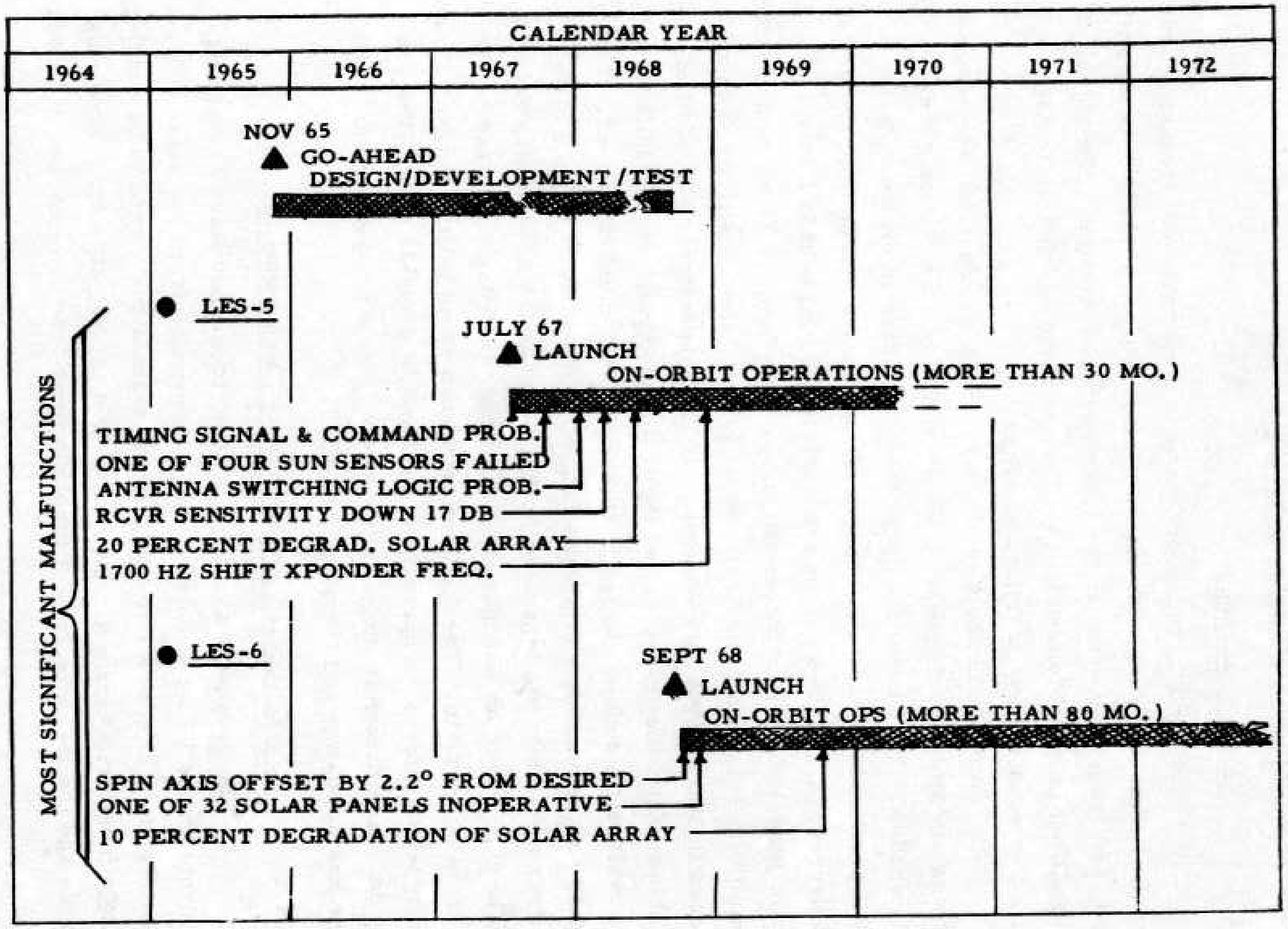



Figure C.4-2. Key Milestones and Events – LES 5 and 6

### C.4.4.4 Satellite Orbital Experience

#### C.4.4.4.1 LES 5 Transponder

The LES 5 transponder makes a conventional frequency translation of the received signal to an intermediate frequency where it is amplified, limited, and translated again to the transmitting frequency. The signal is then amplified in a highly efficient solid state chain and fed to the circularly polarized antenna through a triplexing filter.

Some 8 months after launch, on 18 March 1968, the receiver sensitivity suddenly displayed a 17-dB decrease as LES 5 emerged from the earth's shadow. This sensitivity loss was attributed to an open circuit in the first RF preamplifier that was brought about as the average temperature of LES 5 decreased. (LES 5 average temperature is highest in December and lowest in June.) This hypothesis was borne out when, on 14 November 1968, the sensitivity returned to normal. Subsequently, the sensitivity dropped and recovered in March 1969 and November 1969, respectively. In addition to the problem with sensitivity, one of the two transponder local oscillators (L.O.) exhibited a sudden change in frequency in December of 1968. This oscillator had previously exhibited a long term stability of about 3 in $10^7$. Capacitors similar to the type used in this L.O. have subsequently been observed to experience an abrupt change in value. Although this may be the cause, it is not certain. The L.O. offset shows up as a shift in translation frequency which causes no operational problem to ground users. The sensitivity decrease demands more uplink power from a ground station but in practice has not hampered the usage of LES 5.

#### C.4.4.4.2 LES 5 Command System and RFI Experiments

The command system and radio frequency interference (RFI) experiments on the LES 5 receive timing information from the same source. Immediately after injection into orbit timing signals were seen to be running at a rate almost twice normal. This precluded the satellite from accepting commands and the RFI experiment from stepping frequency at the required rate. Seven hours after ejection the problem spontaneously cleared up for some 5 hours and then reappeared. After several months, of mostly abnormal timing, it became clear that the period of normal timing occurred in the 5-hour interval around that point in the LES 5 orbit when it was opposite the sun. It seems highly likely that the source of the extra pulses is due to an unsuspected coupling from one of the logic systems which acts on earth-sun inputs and changes its state at this point in orbit. This event reinforced the needs for stringent system compatibility tests before launch.

C.4.4.4.3 LES 5 Solar Cell Array

The LES 5 solar cell array was designed to provide power for a 5-year lifetime on the assumption that degradation caused by trapped radiation would amount to a decrease of no more than 5 percent per year in maximum available power. In May 1968, however, the telemetered solar bus voltage was seen to have dropped to 22 V from its launch day value of 30 V. This was the first indication that the LES 5 array was experiencing the same accelerated degradation previously reported on other synchronous spacecraft in early 1968. The best estimate is that the LES 5 array degraded 22 percent in the first year. Had this rate continued, the RF power amplifiers would shortly have shown a marked decrease in output power as their dc converters shut off as a result of low bus voltage.

In addition to difficulties caused by the high degradation rate, the LES 5 solar array experienced an open circuit in one series-connected string on a panel. This had the effect of dropping the available power by an additional 13 percent once per satellite revolution when the affected panel was illuminated by the sun.

C.4.4.4.4 LES 5 Magnetic Stabilization and Antenna Switch System

LES 5 carried an autonomous attitude control system to maintain its spin axis perpendicular to the orbit plane. Orientation is accomplished by a magnetic torquing system. The in-orbit operation of this system was degraded by a failure in one of four sun sensors that trigger the switching action necessary to maintain an inertially fixed magnetic moment. As a result, while the system corrected attitude errors along one axis, it produced small errors along a second, orthogonal axis. In effect this reduced the spin axis correction rate from the expected value of 3/4°/orbit to about 1/4°/orbit. The decrease in rate did not affect the ultimate accuracy, and the LES 5 spin axis was maintained perpendicular to the orbit to within 2° - 2.5°.

This sensor failure also affected the experimental antenna switching logic in that individual sections performed well but the system as a whole did not work. Sufficient information was obtained from the experiment to verify that the system could be used on LES 6 and, with changes to the sun sensor input logic, such a system was flown.

C.4.4.4.5 LES 6 Ejection

Immediately after ejection, telemetered information indicated that LES 6 was not spinning about the expected axis (the symmetry axis of the cylinder, or Y axis in satellite coordinates). Analysis showed that the satellite Y axis was offset from the angular momentum vector by about 2.2°. Another anomaly soon became apparent: one entire solar panel (of 32) was not delivering power. These events are related by the fact that the failed panel lay in the plane defined by the spin axis and the axis of symmetry.

Review of all factors led to the following conclusion: LES 6 either gained or lost weight sometime before it cleared the launch vehicle; to explain the in-orbit satellite unbalance, this weight must have been at least 1 lb. It was further concluded that LES 6 had gained this weight because overall satellite performance and telemetered information could reveal nothing missing and no structural parts of sufficient mass could have conceivably come free. A hypothesis that the offset was there before satellite installation was ruled out. The dynamic balance procedures before launch would have detected < 3 percent of the in-orbit offset. (These were repeated on a LES 6 structural qualification model and their accuracy borne out.)

The resultant spin motion of LES 6 is such that the satellite equatorial plane appears to "wobble" ±2.2°, as the nominal satellite spin axis nutates about the angular momentum vector. This limited the operational use of the autonomous attitude control system.

C.4.4.4.6 LES 6 RF Systems

Two unexpected changes in performance occurred after launch. The power of the transmitted communications signal varied with satellite rotation over a range of 0.9 dB, as a result of a corresponding change in power generated by the solar cells because of the axis tilt and the lack of power from one panel. The second change was a fortuitous decrease from the prelaunch measurements of the power generated in the receiver band by intermodulation between the beacon and communication signals in the common antenna system. This drop in intermodulation power was sufficient to permit simultaneous operation of beacon and communication transmitters with no loss in receiver sensitivity.

The LES 6 beacon and communications signals were chosen so that only a high-order intermodulation product (15th order) fell in the receiver passband. Generation of bothersome intermodulation products at low order had been a problem on LES 5 and it was believed that the higher order product would be acceptably low in power. This was not the case; sporadic noise bursts 10 to 15 dB above the receiver noise level were commonly seen during ground tests. In both satellites, the intermodulation products were generated in spring finger contacts used at the edges of the slot antenna cavities behind the solar panels. Extensive efforts to minimize these products before launch were only partially successful. The spontaneous disappearance of intermodulation products in orbit (in both satellites) led to speculation that some phenomenon such as vacuum welding may have occurred.

C.4.4.4.7 Evaluation of LES 5 and 6 Experience

An annual decrease (from March to November) in LES 5 receiver sensitivity occurred as a result of an open circuit (caused by temperature changes). This anomaly was overcome by using higher uplink power.

A timing pulse anomaly also occurred periodically in LES 5 due to coupling between the logic of satellite subsystems at the point in the orbit when the satellite was opposite the sun. Both of these anomalies indicated the need for more thorough testing, the former for testing over the temperature range expected in orbit and the latter for compatability testing.

A capacitor in LES 5 was suspected of causing local oscillator frequency change approximately 1-1/2 years after launch. Previous experience with capacitors of this type had shown similar behavior.

Several design-related problems were experienced on-orbit. Rapid solar cell degradation in LES 5 was attributed to inadequate radiation protection. A failure in a sun sensor that caused attitude error in LES 5 was solved for LES 6 by a logic design change.

The LES satellites experienced improvement in intermodulation products from in-plant testing to on-orbit operation. Extensive efforts to minimize the products on the ground were only partially successful. However, the product disappeared on-orbit for both satellites. It was speculated that some phenomenon such as vacuum welding may have occurred.

### C.4.5 LES 8 and 9

LES 8 and 9 are the latest* in a series of experimental military communication satellites developed by the MIT Lincoln Laboratory. They will operate with a variety of fixed and mobile terminals using both UHF and K-band** for uplinks and downlinks. A K-band crosslink between LES 8 and LES 9 will also be demonstrated. The communications electronics are all solid state. Two K-band receivers and transmitters are on each satellite, one used with a horn antenna and the other with an 18-in. parabolic reflector. This allows simultaneous communication on both up/downlinks and the crosslink. On-board equipment allows demodulation and remodulation of any signal as well

---

*The LES 7 satellite was intended to have an all solid state, 100-MHz bandwidth, single conversion, X-band repeater. The program was cancelled before the satellite was built.

**Transmission frequencies will be between 36 and 38 GHz.

as routing of signals from any receiver to any transmitter. Additional details are given in Table C.4-6.

LES 8 and 9 are depicted in Figure C.4-3. Both satellites are three-axis-stabilized. The body of each satellite is roughly cylindrical. The K-band antennas are mounted on one end, while two radioisotope thermoelectric generators (RTG), stacked one upon the other, are attached to the other end. The RTGs provide all the electrical power for the satellites; no solar cells are used. The UHF antenna is mounted to the side of the main body. LES 8 and 9 will be launched together on a Titan IIIC booster; the launch is scheduled for the second half of 1975.

Table C.4-6. LES 8 and 9 Technical Details

Satellite

- ~ 850 lb
- Two radioisotope thermoelectric generators, 145 W each initially, 125 W each after 5 years
- Three-axis stabilization, ±0.1° about pitch and roll axes, ±0.6° about yaw axis

Transmitter

- UHF: 240-400 MHz band
- K-band: 36-38 GHz band

Receiver

- UHF: 240-400 MHz band, ~ 4 dB noise figure
- K-band: 36-38 GHz band

Antennas

- UHF: 3 crossed dipoles on a ground plane, 35° beamwidth, ~ 8 dB gain (edge of earth)
- K-band horn, 10° beamwidth, 25 dB gain (on axis)
- K-band dish, 18 in. paraboloid, 1.15° beamwidth, 42.7 dB gain (on axis)

Orbit

- Synchronous, 23° inclination

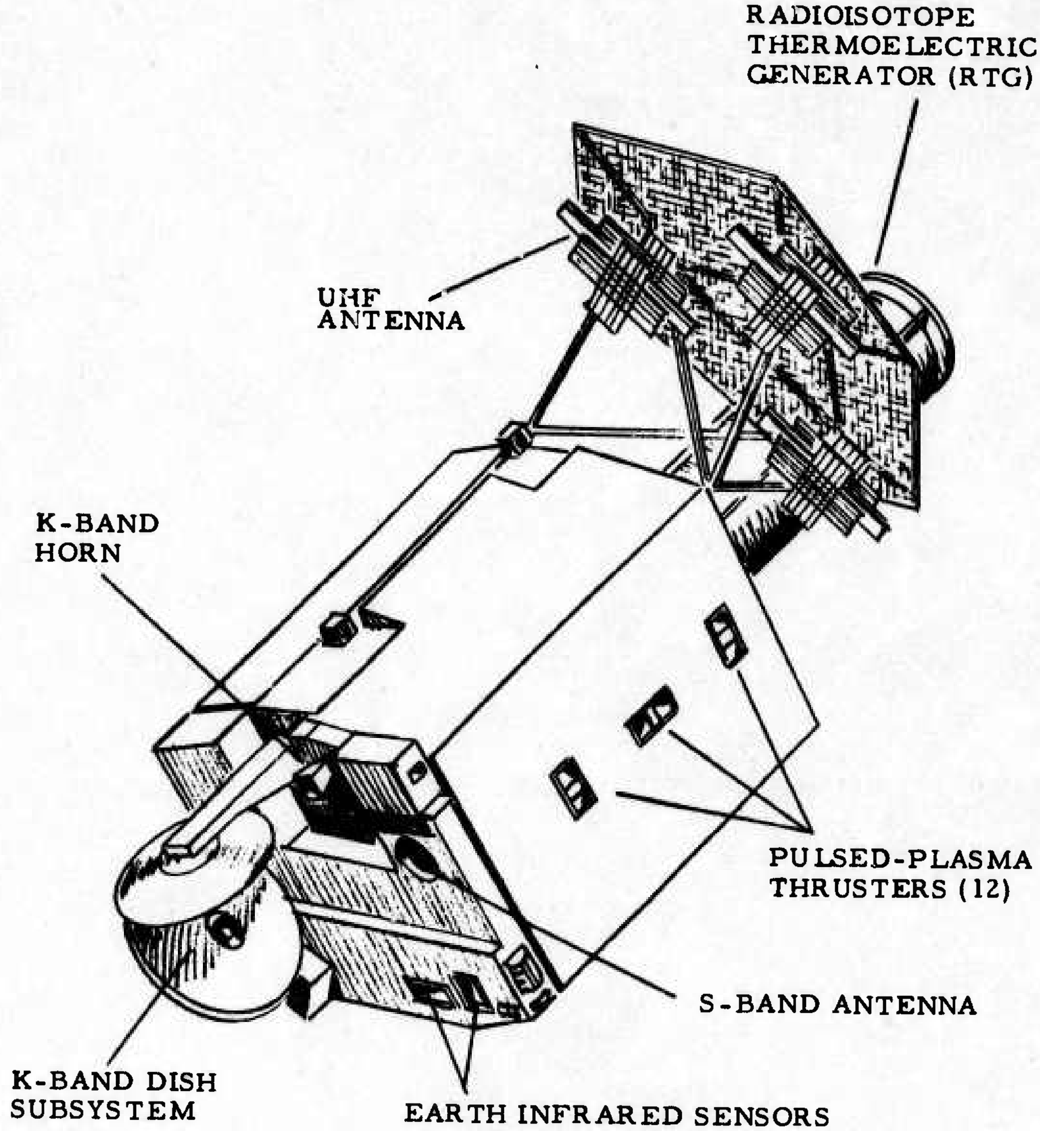
Launch Date

- Late 1975
- Titan IIIC launch vehicle

Developed By

- MIT Lincoln Laboratory

(LES 9 ILLUSTRATED)




LAUNCH DATE

LATE 1975

BOOSTER - T-IIIC

ORBIT - SYNCHRONOUS
$23^{\circ}$ INCLINATION

CHARACTERISTICS

| | | | |
|---|---|---|---|
| WIDTH (ASCENT) | - | 42 | IN |
| WIDTH (DEPLOYED) | - | 54 | IN |
| HT (OVERALL) | - | 120 | IN |
| WT (LIFTOFF) | - | 850 | LB |
| POWER (BOL) | - | 290 | WATTS |

Figure C.4-3. Lincoln Experimental Satellite (LES 8 and 9)

## C.5 DEFENSE METEOROLOGICAL SATELLITE PROGRAM (DMSP)

### C.5.1 Program Summary

The various DMSP models have been designed and fabricated by the RCA Astro Electronics Division at its facility in Princeton, New Jersey. The contracts have been under the direction of the Air Force Space and Missile Systems Organization (SAMSO).

Subsequent to the launch and operation of early models, the first Block 5 satellite was launched on a Thor booster from the Western Test Range in February 1970. Eight additional Block 5A, 5B, and 5C satellites were placed in orbit during the period 1970 through 1974. On-orbit operation of these satellites was conducted for periods of 3 months to 2 years. In 1972, design of the Block 5D configuration was initiated. The first satellite of this design will be launched in late 1975 into a high inclination orbit at 450 nmi altitude.

### C.5.2 Satellite Description

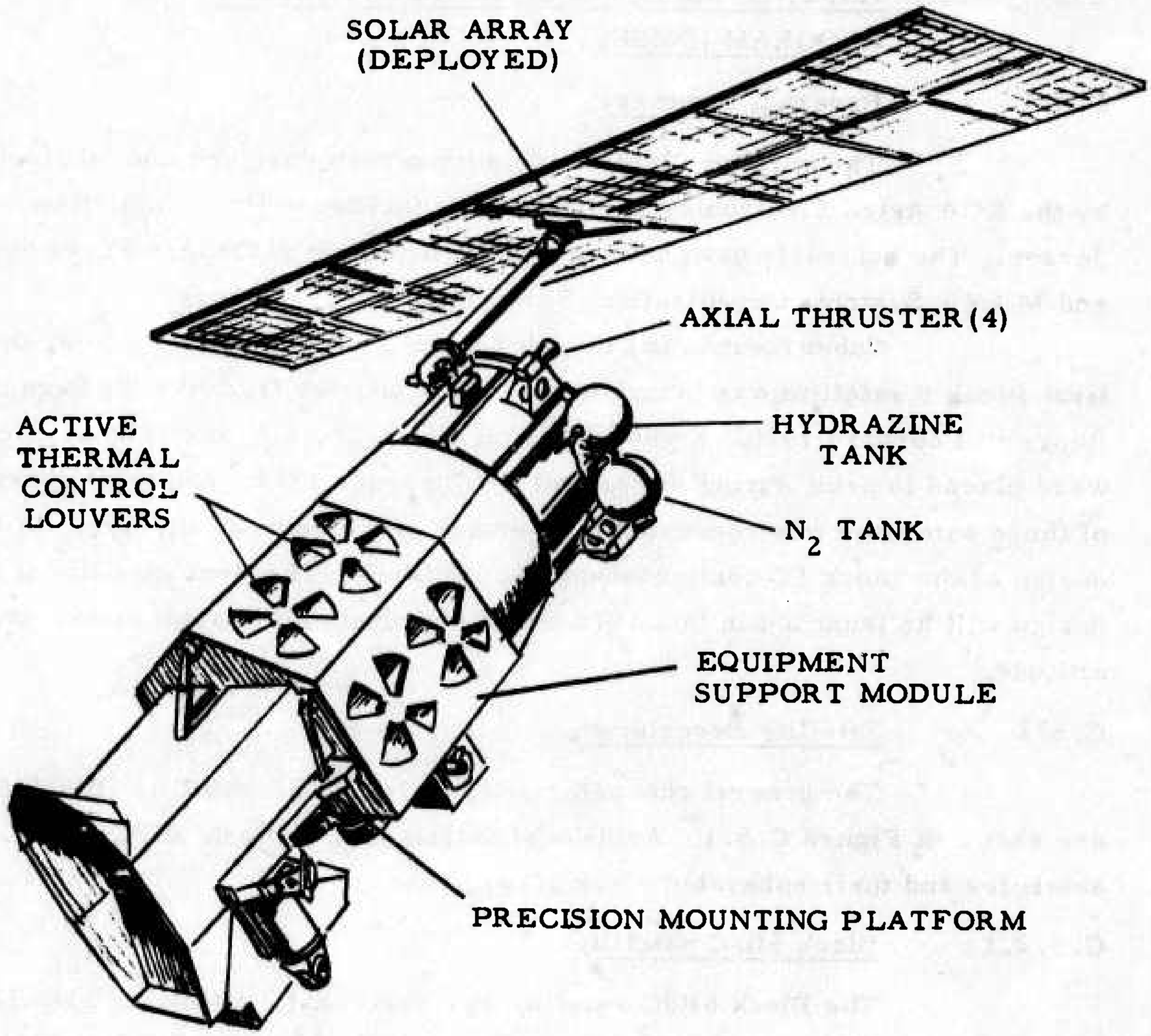
The general characteristics of the DMSP satellite (Block 5D) are shown in Figure C.5-1. Additional details on the Block 5B/C and 5D satellites and their subsystems are given below.

#### C.5.2.1 Block 5B/C Satellite

The Block 5B/C satellite is a three-axis-stabilized vehicle employing a primary sensor package that remains earth-oriented throughout the orbit (nominally "looking" along the local vertical) as well as one or more secondary sensor packages that "look" in various directions relative to the vehicle coordinate system, depending upon the specific data-gathering requirements of each. The total in-orbit satellite weight is approximately 430 lb; of this, approximately 340 lb is allocated to the spacecraft and 90 lb to the sensor packages, which are supplied as government-furnished equipment (GFE).

The primary sensor is a scanning radiometer designed to simultaneously collect both visual and infrared imagery of terrestrial scenes

(BLOCK 5D SATELLITE ILLUSTRATED)




LAUNCH DATE - LATE 1975

BOOSTER - THOR/5D STAGE

ORBIT - 450 NM
98.7 INCLINATION

CHARACTERISTICS

| | | |
|---|---|---|
| WIDTH (ASCENT) | - | 65 IN |
| WIDTH (DEPLOYED) | - | 16 FT |
| HT (OVERALL) | - | 10 FT |
| WEIGHT | - | 5272 LB |
| POWER (BOL) | - | 900 WATTS |
| DESIGN LIFE | | ~2 1/2 YEARS |

Figure C.5-1. Defense Meteorological Satellite Program (DMSP) from a nominal altitude of 450 nmi. It senses radiation continuously both day and night in two spectral intervals by two pairs of detectors. Both of the infrared and one of the visible-spectrum detectors are designed to operate with both day and night illumination conditions. The second visible-spectrum detector, which senses very-high-resolution data, is designed to operate only on or near the daylight side of the day-night terminator.

The satellite is stabilized by a three-axis attitude control system that employs a rotating flywheel for satellite stabilization about the pitch axis. Flywheel momentum and satellite roll/yaw attitude are maintained by magnetic control coils mounted in the solar array "hat." Except for the flywheel, the satellite, including the spin and attitude control coils located inside the solar array hat, rotates at one revolution per orbit to maintain earth orientation with the primary sensor and the data transmission antenna located on the earth-facing side.

A structural baseplate supports all components and transmits stresses to a three-point booster mounting system. A 12-sided solar array "hat" covers the electronic components (except for the primary sensor unit and selected secondary sensor units) and provides mounting surfaces for the solar cells, which are the main source of power. The primary sensor package and the S-band antennas are located on the earth-oriented side of the hat. The flywheel and pitch sensor are located on the underside of the baseplate.

The spacecraft is designed to support the data-acquisition functions of the GFE sensor packages. To fulfill this function, the spacecraft comprises eight functional subsystems:

a. Primary and secondary data-processing subsystems;
b. Communications subsystem;
c. Attitude control and vehicle dynamics subsystem;
d. Controls subsystem;
e. Power subsystem;
f. Structure; and
g. Thermal control.

C.5.2.1.1 Primary and Secondary Data Processing Subsystems

The primary and secondary data-processing subsystems process all data signals from the payload (primary and secondary packages) for real-time and/or remote transmission to ground stations. For real-time transmission, the processing consists of converting the primary payload data signals to a digital format, encrypting the digital data if desired, phase-shift-modulating the data, and transmitting the modulated signal to the ground stations via an S-band antenna. Primary payload data, together with associated flutter and sync signals, are processed and transmitted simultaneously.

To perform their functions, the primary and secondary data processing subsystems include three magnetic tape recorders, a five-channel frequency-multiplexer, a secondary subcarrier oscillator, units for analog-to-digital conversion and formatting, encryption equipment, a solid-state S-band transmitter, a modulator-driver unit/traveling wave tube amplifier (TWTA) transmitter, and two S-band antennas.

C.5.2.1.2 Communications Subsystem

The communications subsystem provides all communications (except primary and secondary data) to and from the satellite. Specifically, this comprises a command channel and a tracking and telemetry channel.

The command channel consists of a VHF antenna and a receiver-demodulator unit. The antenna is made up of four monopole elements attached to the underside of the baseplate. The receiver is an AM type that detects the subcarrier containing the command messages. The subcarrier is then demodulated to recover the frequency-shift-keyed commands. The resulting digital signals are supplied to the controls subsystem, which interprets them as addresses, commands, or data.

The tracking and telemetry channel furnishes an RF carrier, in the UHF band, that is used by the ground stations to track the satellite. A single whip antenna attached to the top of the solar array hat radiates this carrier, which is generated by the tracking/acquisition transmitter.

C.5.2.1.3 Attitude Control and Vehicle Dynamics Subsystem

The attitude control and vehicle dynamics subsystem includes all equipment for attitude control, nutation control, and pitch stabilization. The payload is the primary source of attitude data; however, pitch sensor data can be used, with lower accuracy. The Block 5B/C satellite includes a three-axis stabilization system in which the satellite is spin-stabilized by a flywheel rotating at 150 rpm. The remainder of the satellite is "spun" at one rotation per orbit to earth-orient the sensor.

Nutation is controlled by a passive liquid damper, and attitude (roll and yaw) is controlled magnetically by two magnetic bias coils (each) associated with a 12-position switch by which coil current can be varied) and by an attitude control coil. Pitch-axis control is provided by a servo system motor that exerts a torque between the flywheel and the body of the satellite.

A coil is provided to exert, by interaction with the earth's magnetic field, a torque on the satellite body. This torque is used, when required, to replace angular momentum which is lost due to hysteresis, eddy currents, etc.

C.5.2.1.4 Controls Subsystem

The controls subsystem contains the logic associated with receiving and executing commands and stored programs, and also the circuitry required to generate timing and control signals that govern the proper functioning of all other satellite components.

Commands and program data are received from the command receiver-demodulator as digital signals. A high degree of rejection is provided against the spacecraft accepting extraneous signals as commands. An address code is provided to guard against accepting commands intended for another spacecraft.

Commands may be sent in real time for immediate execution, or they may be sent, together with time tags, as program data to be stored in the main memory for execution at a later time. Up to 128 commands may be stored.

Two additional memory units are provided. The first of these is the orbit memory, which normally contains a small set of control commands (up to 31) executed once each orbit. The second is the data memory, which contains 184 decimal digits of data in BCD format. The stored data are read out, together with the primary sensor data, in the satellite's direct readout modes. These data may be used to provide data users in remote locations with information on the satellite ephemeris or other pertinent data.

The controls subsystem also contains a timing chain that provides all the timing for internal operations and also for a number of other components such as the sensors and the recorders.

#### C.5.2.1.5 Power Subsystem

The power subsystem provides all electrical power, both regulated and unregulated, for satellite operation. The primary power source is a solar cell array mounted to the top and 11 of the 12 sides of the "hat" structure. During the daytime portions of the orbit, the array output power is divided between the satellite loads and the battery and its charge control system, with any excess going to the shunt dissipator. During the night portions of the orbit, power for satellite operation is furnished from a 15-cell NiCd battery. The battery also furnishes power to augment the solar array output during peak load periods.

#### C.5.2.1.6 Structure Subsystem

The structure subsystem includes the baseplate, the solar array structure, the separation system, and all of the various brackets and mechanical devices needed for component mounting. The baseplate (the primary load-carrying structure) is a flat circular platform of 52 in. diameter, machined from a magnesium forging. The solar array structure is a 12-sided tapered polyhedron fabricated from aluminum honeycomb with aluminum skins.

#### C.5.2.1.7 Thermal Control

Satellite thermal control is achieved by a combination of passive and active techniques. Heat flow between the inside surfaces of the solar array structure and the components mounted atop the baseplate is minimized by an aluminized Mylar mirror stretching across the structure above the components and by insulating blankets. The external surfaces of the solar array structure are finished (except for the solar cells) to act as a heat radiator, and the array structure is thermally decoupled from the baseplate. In addition, thermal blankets surround the GFE sensors.

Active thermal control is provided by a group of louvers located in the annular area between the flywheel and the baseplate periphery. Each louver is a "sandwich" of aluminized Mylar "skins" bonded to a polyurethane foam center, and is opened and closed automatically by a bimetallic spring mounted to the baseplate.

C.5.2.2 Block 5D Satellite

The spacecraft (see Figure C.5-1) features virtually complete "hands off" operation, utilizing redundant CMOS (complementary metal oxide semiconductor) data processors to achieve precise automatic command and control of all spacecraft functions, including high-accuracy attitude determination and control for accurate sensor "pointing." A probability of survival of approximately 0.8 over the mission design life of 2-1/2 years is achieved by selective redundancy, with automatic switchover in the event of a failure.

A wide variety of GFE sensors are carried aboard the spacecraft to produce high-quality visible and infrared cloud cover imagery as well as vertical temperature profile measurements. Selected areas of the earth can be viewed with a resolution of 0.3 nmi in both the visible and infrared spectra. For complete global coverage, 1.5-nmi-resolution visual and infrared imagery can be obtained by the onboard primary sensors. The special sensor complement can be varied according to specific needs for each satellite, including such sensors as a vertical temperature profile sounder. Imagery and secondary data can be stored on recorders for delayed transmission or relayed directly to data acquisition centers via the spacecraft data transmitters.

Additional details on major satellite subsystems are given below.

#### C.5.2.2.1 Structure

A 10-ft magnesium and aluminum structure serves as the fixed surface for mounting and connecting components and for linking the spacecraft to the second stage. It has been designed especially to withstand launch, ascent, and orbital environments, and to maintain alignment accuracy.

The primary sensor and other components requiring high-accuracy alignment are mounted to the Precision Mounting Platform (PMP). It consists of an aluminum "egg-crate" structure measuring 40 by 28 by 2-3/4 in. Designed to support 200 lb, the platform is attached to the Equipment Support Module by two ball-joint supports at the base, and two thin-wall aluminum struts with ball-joint mountings at each end.

Mounted to the Equipment Support Module (ESM) are the S-band turnstile antennas, portions of the GFE sensor payload, and various electronic components. The sensors and antennas are attached to an earth-facing panel; the electronic components are located on the inner surfaces. The polygon-shaped module frame is covered by bonded honeycomb aluminum panels. The module frame is 39 in. high and fits inside a 44-in.-diameter cylinder. An aluminum truss that connects the module to the Reaction Control Equipment Support Structure is 21 in. high.

The cylindrical Reaction Control Equipment Support Structure (RSS) houses the third-stage motor as well as supporting the reaction control equipment on its outer periphery. The battery packs and battery charge assembly, together with their thermal controllers, the solar array boom, and the S-band quadrifilar antennas are also mounted on the RSS.

#### C.5.2.2.2 Reaction Control

The reaction control equipment (RCE), a pressurized nitrogen and hydrazine system, provides three-axis steering following booster separation. It consists of four spherical titanium propellant tanks, two of which contain 4.6 lb of nitrogen and the other two, 32.5 lb of hydrazine. These tanks feed two pairs of hydrazine thrusters, each with the capability of generating either 55 or 165 lb of thrust, and eight nitrogen thrusters, each generating 2 lb of thrust.

C.5.2.2.3 <u>Thermal Control</u>

Both active and passive thermal control are employed to maintain correct temperature of all components and electronics. One of two types of active thermal control employs pinwheel-like louvers, seven of which are located on the ESM. Each consists of a four-section louver blade rotating on a bimetallic coil. The fiberglass blade is covered by aluminized Kapton film. The second type of active thermal control uses rectangular-shaped louvers that are made of Urethane foam covered by aluminized Kapton film and reinforced with fiberglass quill.

Various finishes, materials, blankets, and shields are employed for passive control.

C.5.2.2.4 <u>Attitude Determination and Control</u>

An accurate (better than $0.1^{\circ}$), three-axis attitude determination and control subsystem permits precise pointing of the sensor payload located on the PMP. Three onboard orthogonal gyroscopes measure short term changes in attitude. A star sensor provides the data necessary to compensate for gyro drift. An onboard processor stores ephemeris data and computes the satellite attitude. To enhance pointing accuracy, extensive star catalogs and ephemeris tables are periodically transmitted to the spacecraft from the ground. A back-up gyroscope is available in the event of failure of any of the other three. Attitude control is provided by three reaction wheels in an active closed-loop configuration (with a fourth for back-up) and by magnetic coils for unloading excess momentum.

In the event of failure of the inertial measurement unit (IMU) or star sensor on the primary system, lower accuracy (better than $0.2^{\circ}$) three-axis attitude determination and control is available. It is provided by earth horizon and sun position sensors. When the primary system is working normally, the back-up sensors operate in the monitoring mode and represent an additional data source.

### C.5.2.2.5 Power

A deployable, coplanar, sun-tracking solar array (SA), canted at 28° to the spacecraft pitch axis, provides power when the spacecraft is in sunlight. The eight-panel honeycomb array is covered with 10,560 silicon solar cells. The array delivers at least 325 W (worst case, end of life). The eight panels, each measuring 2 by 6 ft, are connected by spring-loaded locking hinges to give a total array area 6 by 16 ft.

A 17-cell rechargeable NiCd battery, rated at 30 A-hr, provides power at night and during peak daylight loads. The 1.25-V cells are connected in series to produce a nominal 21.25 V.

### C.5.2.2.6 Communications and Telemetry

The communications subsystem includes five S-band spacecraft-to-ground links, three for data, two for telemetry, and an S-band ground-to-spacecraft command link. Separate antennas are provided for each link. The telemetry links can also be used as back-up for the data links if necessary. The telemetry subsystem has 256 analog channels and 160 bi-level discrete channels.

Each data link includes an S-band transmitter, filter, and S-band dipole antenna for PM transmission of digital NRZ data at 1.024, 1.3312, or 2.6624 Mbps. The high-power link provides RF power of 11 W at 2267.5 MHz; the low-power link and direct data link each provide 5.5 W RF output at 2207.5 MHz and 2252.5 MHz, respectively. Minimum power at ground station antenna is -133 dBm for the low power and direct data links.

Each telemetry link includes a transmitter, filter, and antenna for FM or PM transmission of analog data or bi-phase digital data at 2 or 10 kbps in orbit (60 kbps during ascent). Either link can be used as a back-up for either the low-power or direct data links at full data rates. The transmitters and filters are essentially identical to the data link units. One of the two antennas is also identical to those of the data links; the second is a quadrifilar pair that gives an omni-directional, right-hand circular polarized pattern. Minimum power at the ground station is -152 dBm for either link. During ascent, telemetry data are processed at 60 kbps. In orbit, telemetry speed is selectable at 2 kbps (slow PCM) or 10 kbps (fast PCM).

The SGLS-compatible command link operates at 1791.748 MHz. It includes an omnidirectional quadrifilar pair antenna, a filter network, and a fully redundant receiver-demodulator unit (RDU). The command data are frequency-modulated at 1 kbps.

#### C.5.2.2.7 Command and Control

The all-digital command and control subsystem provides guidance signals during ascent, and controls the spacecraft attitude and operating modes while in orbit. On-orbit control may be handled by commands and data from the ground or from other on-board subsystems. The system includes redundant central processing units (one containing the ascent load program during ascent), a high stability redundant oscillator, and interface circuitry.

### C.5.3 Key Events and Milestones

Significant milestones of the DMSP satellites are given in Figure C.5-2. Major on-orbit malfunctions for Block 5A/C and 5B/C spacecraft are also indicated.

### C.5.4 On-Orbit Malfunctions

A summary of malfunctions experienced on the Block 5B/C satellites is given in Table C.5-1.

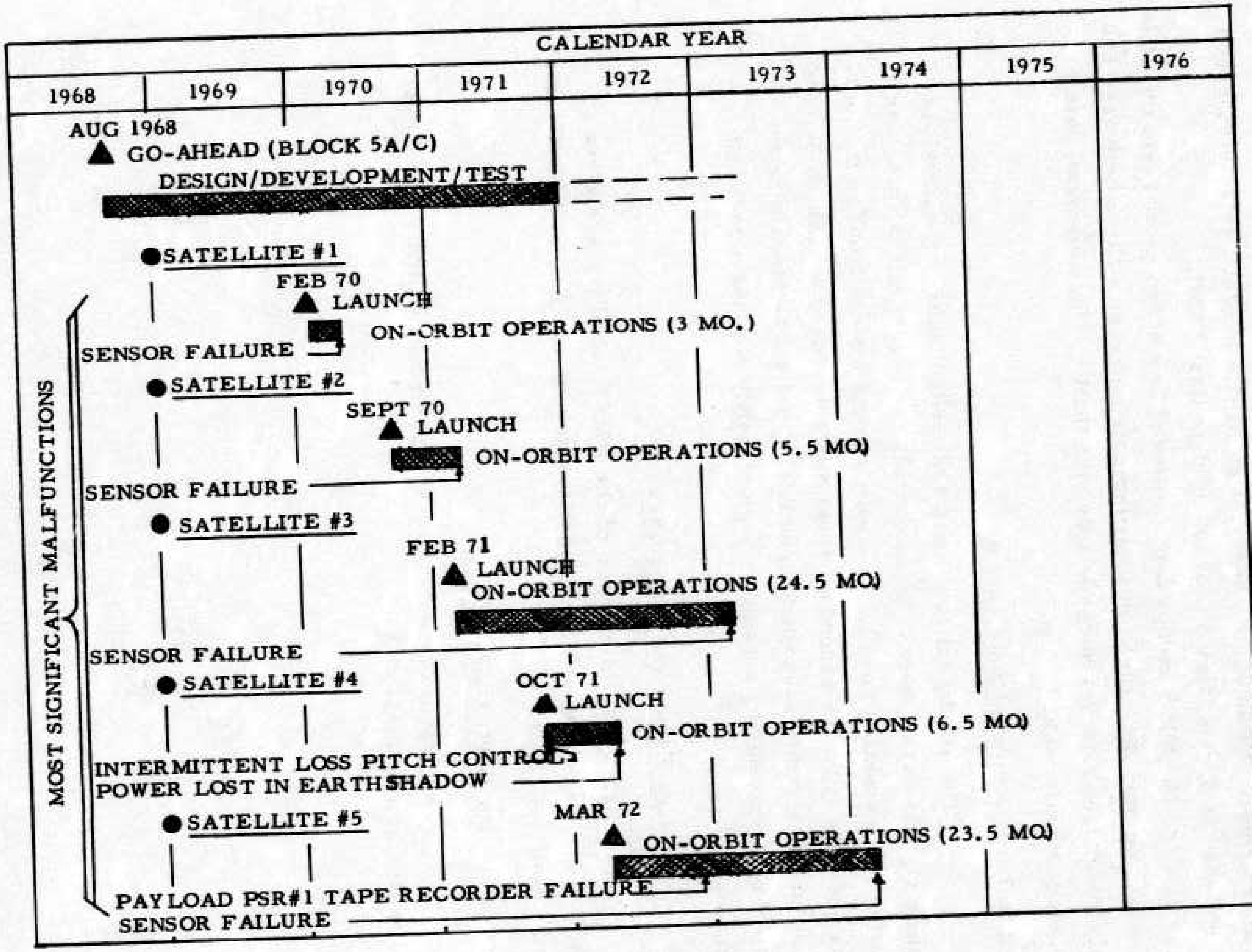



Figure C.5-2. Key Milestones and Events - DMSP

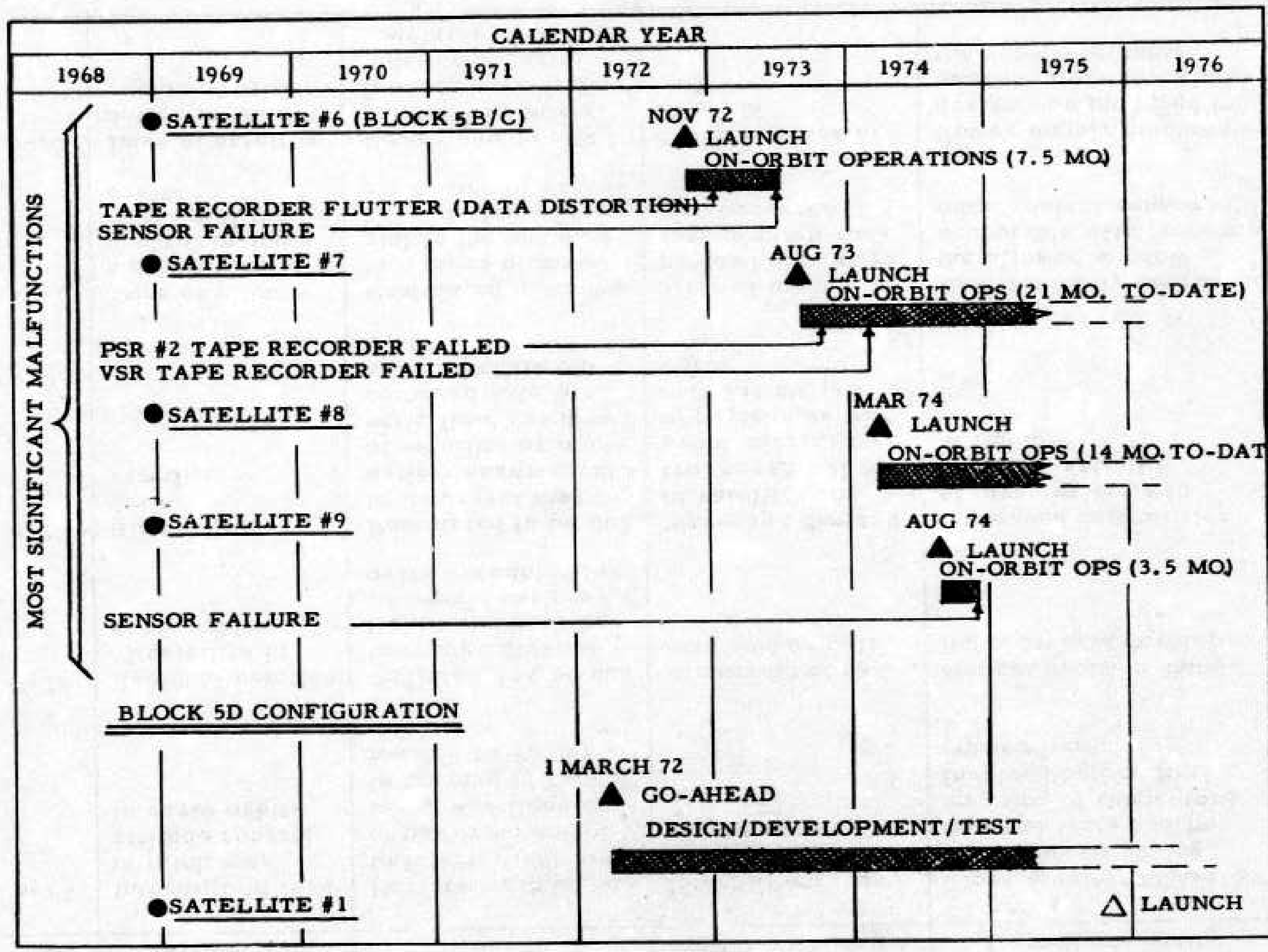



Figure C.5-2. Key Milestones and Events - DMSP (Cont.)

Table C.5-1. Major On-Orbit Malfunction Summary - Block B/C

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 8-73 | Intermittent loss of pitch axis attitude control in early orbits | Postulated to be due to intermittent loss of power on one of the power lines due to opening of connection on solder joint | Intermittent loss of usable payload sensor data | Close monitor of system performance during thermal tests and inspection of connectors involved before final connections |
| 8-73 | Residual nutation of satellite in early orbits | Postulated to be due to energy dissipation in the bearing assembly not properly accounted for | Distortion of payload sensor data | Manual nutation damping technique developed |
| 8-73 | High attitude drift of satellite | Postulated to be due to incorrect magnetic measurements of satellite or magnetic field changes on board satellite after satellite on-orbit | Increased demand of satellite control personnel to avoid correction of payload sensor data for attitude error | Increased capabilities of torquing coils to maintain satellite attitude |
| 10-71 | Loss of pitch axis attitude control in some regions of the orbit | Postulated to be due to a piece of material in the scanned field of view of the pitch control sensor | Loss of usable payload data when loss of pitch control occurred | Electronically decreased the allowed window of acceptable data from the pitch control sensor |
| 5-72 | Loss of satellite power in "dark" region of orbit | Power supply electronics was postulated to have malfunctioned due to regulator noise | Complete loss of mission | Power supply electronic design was modified to provide more margin in the boost regulator |

Table C.5-1. Major On-Orbit Malfunction Summary - Block B/C (Cont.)

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 1-73 | Tape recorder VSR flutter | Postulated to be due to degraded bearing in capstan drive or stickiness between heads and tape causing "hunting" of motor | Distortion of payload sensor data from recorder | Replacement of motor and retest to verify the non-existence of the typical signature of "hunting" motor |
| -- | Tape recorder failed to operate (several failures of this type) | Some postulated to be due to loss of drive (belt or pulley was loose), some postulated to be due to failure in power supply and/or logic | Loss of payload sensor data from recorder | Replacement of suspect parts; close inspection of test data from testing the recorder and/or components involved in postulated cause of malfunction |

## C.6 INITIAL DEFENSE COMMUNICATIONS SATELLITE PROGRAM (IDCSP)

### C.6.1 Program Summary

The IDCSP satellites were designed and fabricated by the Space and Reentry Systems Division of the Philco-Ford Corporation at Palo Alto, California. The program was under the direction of the Air Force Space and Missile Systems Organization (SAMSO) with General Systems Engineering/Technical Direction performed by The Aerospace Corporation.

The IDCSP Initial Definition Phase started in 1962 and the production contract in October of 1964. Five launches placed four sets of satellites in sub-synchronous, nearly equatorial orbit; the second set was lost when the launch vehicle fairing failed. The IDCSP satellites were placed in orbit (along with other experimental satellites in several instances) by Titan IIIC boosters as follows:

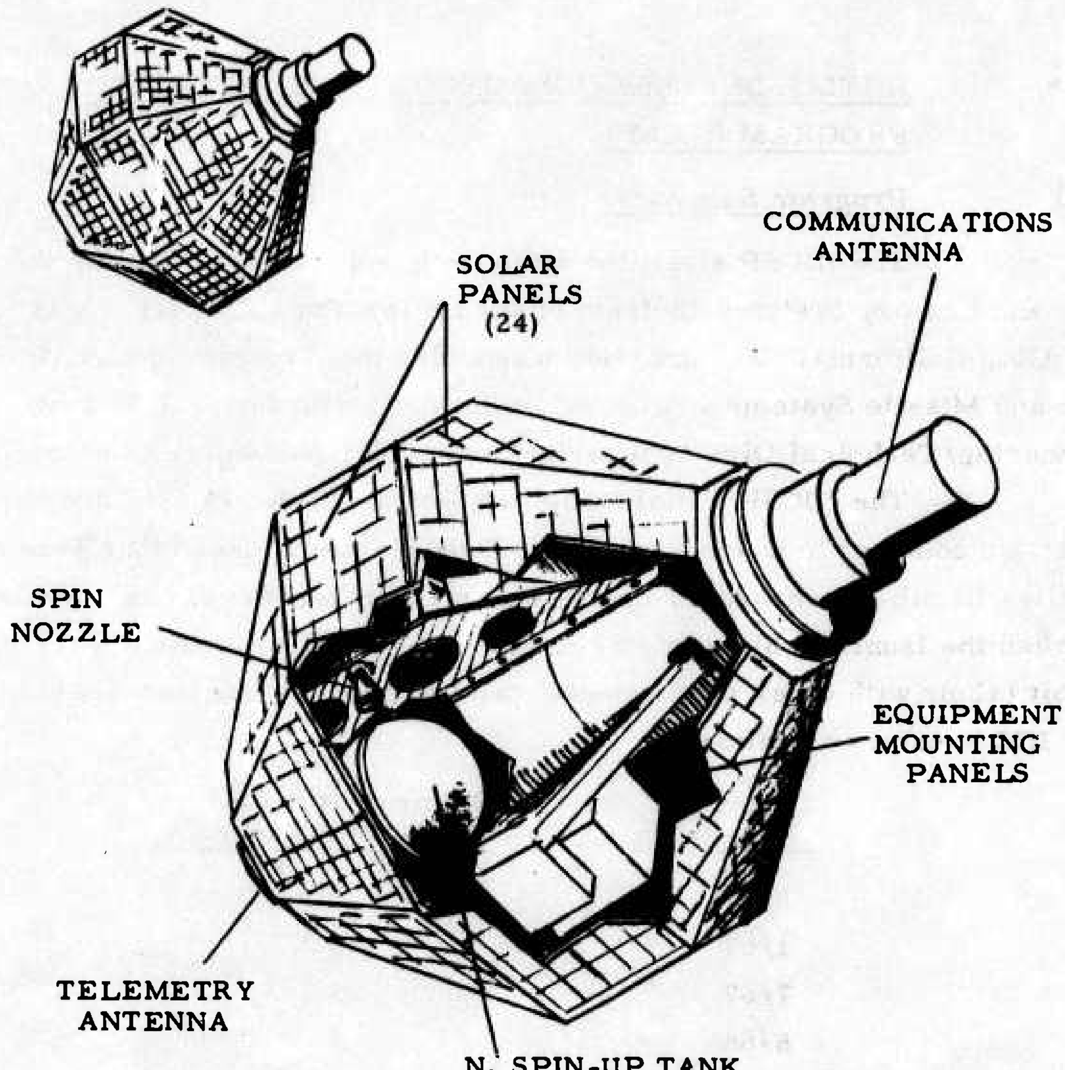
| Launch Date | Number of IDCSP Spacecraft/Launch |
|---|---|
| 6/66 | 7 |
| 1/67 | 8 |
| 7/67 | 3 |
| 6/68 | 8 |

### C.6.2 Satellite Description

An overview of the satellite is shown in Figure C.6-1. The major items comprising the satellite subsystems are listed in Table C.6-1.

### C.6.3 Key Milestones and Events

The significant program milestones are shown in Figure C.6-2, which also lists the most significant malfunctions that occurred during IDCSP orbital operations. This information was obtained from References C.6-1 and C.6-2.




LAUNCH DATES

| | | |
|---|---|---|
| #1-7 | - | 16 JUNE 1966 |
| #8-15 | - | 18 JAN 1967 |
| #16-18 | - | 1 JULY 1967 |
| #19-26 | - | 13 JUNE 1968 |

| | | |
|---|---|---|
| BOOSTER | - | TITAN III-C |
| ORBIT | - | 18,000 NM EQUATORIAL |

CHARACTERISTICS

| | | |
|---|---|---|
| DIAMETER | - | 36 IN |
| HEIGHT | - | 32 IN |
| WEIGHT | - | 100 LB |
| POWER (BOL) | - | 40 WATTS |
| DESIGN LIFE | - | 1.5 YEARS |

Figure C.6-1. Initial Defense Communications Satellite Program (IDCSP)

Table C.6-1. IDCSP Satellite Description

**General**

- Polyhedron spinner (150 rpm)
- 36 in. diameter by 32 in. high
- 100 lb
- 1.5 year MTTF
- Nitrogen fed spin-up jets (one time)
- Passive thermal control

**Structure**

- Central cylinder
- Frame
- Frame to cylinder panels

**Electrical Power**

- Body-mounted solar array (8000 cells)
- 40 W BOL power
- No batteries
- Distribution and power control circuitry

**Communications Subsystem**

- One double-conversion repeater (20 MHz bandwidth)
- X-band frequency
- Two redundant TWTs (3 W)
- 7 dBW ERP maximum
- Two fixed biconical horn antennas (1 transmit, 1 receive)
- Redundancy control unit (for automatic switching to redundant TWTA)

**Telemetry**

- Telemetry generator
- Transmitter (UHF)
- Sun angle sensor
- Antenna

**Timer**

- 6-year timer (-0, +6 mo)
- Shutoff comm. and TLM

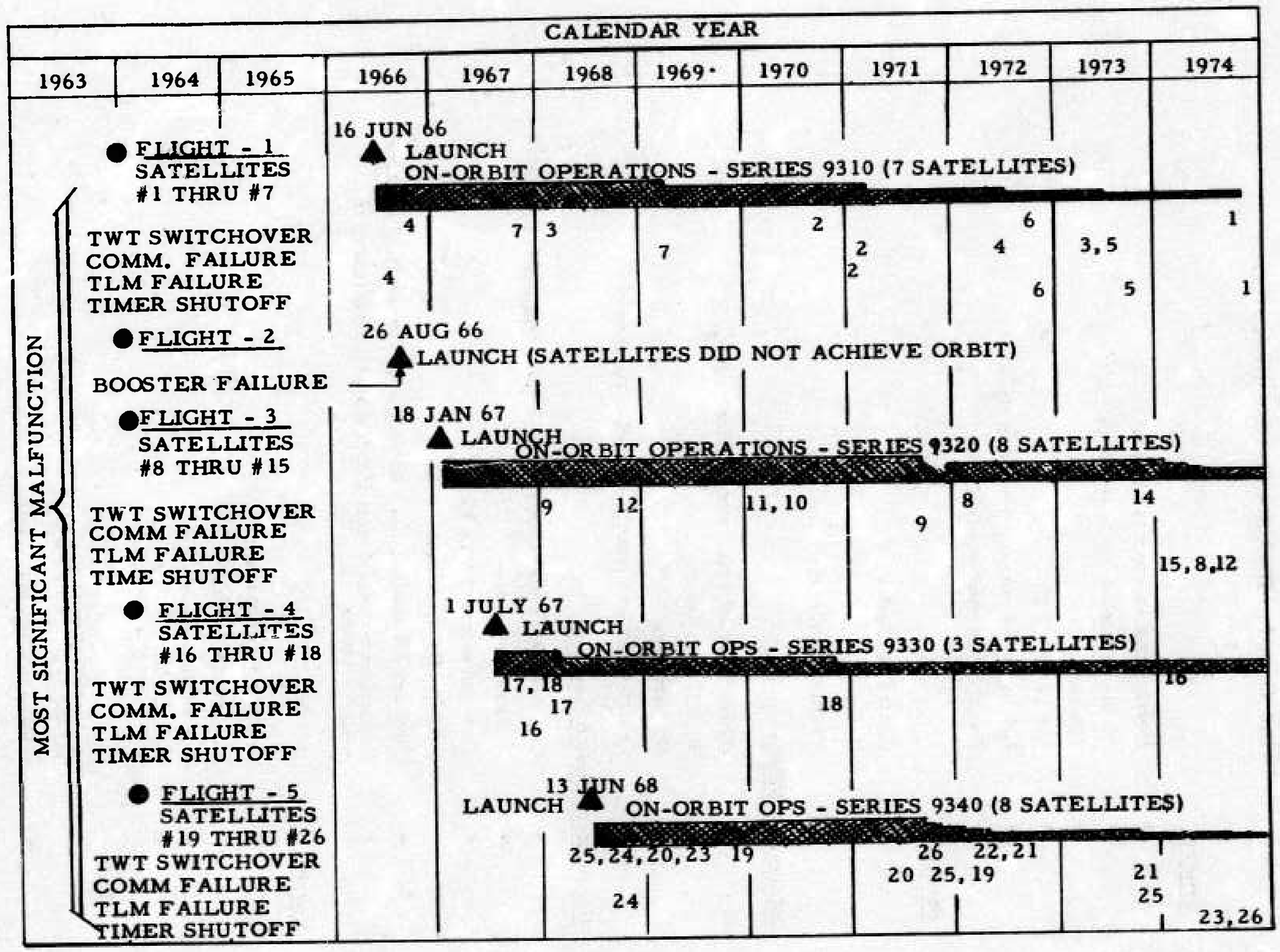



Figure C.6-2. Key Milestones and Events — IDCSP

## C.6.4 On-Orbit Satellite Experience

Failures in the X-band transponder up-converters were attributed to a blind tuning screw set too close to the inner conductor, resulting in an RF short when subjected to thermally induced mechanical distortion. This problem was avoided in the last launch satellites by tighter process controls that guaranteed an adequate gap through use of gauges.

Premature switching of TWTAs was attributed to open printed circuit paths in the Redundancy Control Unit (RCU). This problem was chronic in the last launch of satellites and is thought to be related to the many instances of intermittency ranging from a few hours to many months. Most of these intermittencies occurred during or after eclipse periods or near the winter solstice, indicating thermal stress as the most likely forcing function. Susceptibility to the thermal stressing was thought to be introduced by a multilayer printed circuit mother board assembly problem. Assembly jigs and tooling used on the last launch of satellites were quite worn from previous use, resulting in misalignment of many of the 1200 pins that penetrated one or more of the five layers. Thus, mechanical stresses were built in during assembly and aggravated later by thermally induced stresses.

The several early telemetry failures were also attributed to a thermal stressing problem in the Power Control Unit (PCU) multilayer mother board.

On-orbit malfunctions of IDCSP satellites are summarized in Tables C.6-2 through C.6-5.*

*Note: in the TWTA column of these tables, I signifies primary and R redundant; the TWTA manufacturers are also indicated, WJ for Watkins-Johnson and E for Eimac (Varian).

Table C.6-2. On-Orbit Malfunction Summary – Satellites 9311 through 9317, Launched June 1966

| Telemetry Characteristics | | | Communication Characteristics | | | |
|---|---|---|---|---|---|---|
| Sat. ID. | Performance | Comments | Performance | TWTA I | TWTA R | Comments |
| 9311 | Timed out | | Timed Out | WJ Switched 5-10-74 | E | Suspected switching cause is RCU failure or random failure in TWTA. Timer switched satellite off 6-24-74. |
| 9312 | Ceased | Telemetry ceased 2-8-71 | Failed | E Switched 8-27-70 | WJ | Suspected switching cause is RCU failure or random failure in TWTA. Ceased 2-9-71; cause attributed to TWTA failure. |
| 9313 | Nominal | TLM signal strength low, but is no problem to user. | Failed | E Switched: 2-23-68 | WJ | Switching caused by tube heater opening in eclipse season, but power output was declining prior to this. Failed 3-13-73. |
| 9314 | Ceased* | TLM output power degraded during L+2 months | Failed | WJ Switched: 9-12-66 | E | Suspected switching cause is RCU failure during eclipse. Classified as failed unit 3-20-72 due to TWTA degradation causing low ERP. |
| 9315 | Timed Out | | Timed Out | WJ | E | Timed out 8-27-73. |
| 9316 | Timed Out | | Timed Out | E Switched: between 8-18-72 & 8-21-72 | WJ | Suspected cause of switching is TWTA degradation. Timer switched satellite off between 11-24-72 and 11-27-72. |
| 9317 | Nominal | | Ceased | WJ Switched: 9-29-67 | E | Switching caused by failing tube and up-converter output: output power did not recover due to low up-converter output. Classified as failed unit due to low ERP 5-1-69. Ceased 5-6-71. |

* History of intermittent performance.

Table C.6-3. On-Orbit Malfunction Summary – Satellites 9321 through 9328, Launched January 1967

| | Telemetry Characteristics | | Communication Characteristics | | | |
|---|---|---|---|---|---|---|
| | | | | TWTA | | |
| Sat. ID. | Performance | Comments | Performance | I | R | Comments |
| 9321 | Timed out | TLM signal strength low, but is no problem to user. | Timed out | E | WJ | Suspected cause of switching is TWTA degradation. Timed out 4-3-74. |
| | | | | Switched: 6-5-72 | | |
| 9322 | Nominal | | Failed | WJ | E | Suspected switching cause is RCU failure. Classified as failed unit 12-1-71 due to low ERP. |
| | | | | Switched: 2-15-68 | | |
| 9323 | Nominal | | Nominal | E | WJ | Suspected switching cause is TWTA failure. TWTA operation degraded continuously since Sept. 1969. |
| | | | | Switched: 3-7-70 | | |
| 9324 | Nominal | | Nominal | E | E | Suspected switching cause is TWTA which degraded continuously since launch. Redundant TWTA ERP has degraded continuously since March 1973. |
| | | | | Switched: 1-24-70 | | |
| 9325 | Timed out | | Timed out | E | WJ | Low up-converter drive to degraded TWTA, resulting in loss of output power, caused tube switching. Timed out 4-4-74. |
| | | | | Switched: 10-11-68 | | |
| 9326 | Nominal | | Nominal | WJ | E | |
| 9327 | Nominal | | Nominal | WJ | E | Suspected switching cause is TWTA failure |
| | | | | Switched: 12-3-73 | | |
| 9328 | Timed out | | Timed out | E | WJ | Timed out 3-27-74. |

Table C.6-4. On-Orbit Malfunction Summary — Satellites 9331 through 9333, Launched July 1967

| | Telemetry Characteristics | | Communication Characteristics | | | |
|---|---|---|---|---|---|---|
| | | | | TWTA | | |
| Sat. ID. | Performance | Comments | Performance | I | R | |
| 9331 | Ceased | All analog telemetry ceased; no transmission since 9-19-67. | Nominal | E Switched: 3-11-74 | WJ | Classified as a failed unit 1 Dec 1973 due to low ERP. Suspected switching cause is degraded TWTA ERP or tube failure. |
| 9332 | Nominal* | Telemetry signal exhibits occasional low signal strength as of 9-22-70. | Failed | E Switched: 7-1-67 | WJ | The switching cause is failure of the up-converter. Not usable during eclipse season due to low output power. Classified as a failed unit 3-19-68. |
| 9333 | Nominal* | Telemetry signal exhibits occasional low signal strength as of 11-30-70. | Failed | E Switched: 9-24-67 | E | Suspected switching cause is RCU failure during eclipse. Classified as a failed unit 11-30-70 due to TWTA degradation causing low ERP. |

*History of intermittent performance.

Table C.6-5. On-Orbit Malfunction Summary — Satellites 9341 through 9348, Launched June 1968

| | Telemetry Characteristics | | Communications Characteristics | | | |
|---|---|---|---|---|---|---|
| | | | | TWTA | | |
| Sat. ID. | Performance | Comments | Performance | I | R | Comments |
| 9341 | Nominal* | Inoperative 10-16-68 to 2-23-69 and 11-13-70 to 2-18-71. Telemetry ceased 12-25-71; recommenced 2-16-72. | Ceased* | WJ<br>Switched: 9-17-69 | E | All signals ceased 10-16-68, recommenced 2-23-69. Cause is intermittent RCU/PCU circuits. RCU suspected as cause of TWTA switching. All signals ceased 11-13-70. Recommenced 2-18-71. All signals ceased 12-25-71; recommenced 2-16-72. All signals ceased 3-4-72. |
| 9342 | Nominal* | Inoperative 6-21-71 to 8-25-71. | Failed | E<br>Switched: 9-2-68 | WJ | Suspected switching cause is RCU failure during eclipse but low local oscillator power and corresponding low communications output lasting one day occurred on 9-12-68. Classified as non-operational due to low ERP 5-9-71. All signals ceased 6-29-71, recommenced 8-25-71. |
| 9343 | Nominal* | Inoperative 1-24-72 to 2-18-72. | Failed | WJ<br>Switched: 6-5-72 | E | All signals ceased 1-24-72; recommenced 2-18-72. Initial TWTA parameters indicated nominal tube performance at time of switching. Suspected cause of switching is failure in associated electronics. Failed 8-22-73 due to low ERP. |
| 9344 | Nominal* | Inoperative 9-18-68 to 9-19-68, 11-21-68 to 2-24-69, 8-3-70 to 8-27-70, and 12-17-70 to 2-1-71, Telemetry ceased 12-1-71; recommenced 2-16-72; ceased 4-22-73; recommenced 8-17-73. Ceased 5-9-74; recommenced 5-22-74. | Nominal* | WJ<br>Switched: 3-13-72 | E | Initially all signals ceased 9-19-68, recommenced 9-19-68. Cause is intermittent RCU/PCU circuits reconnecting solar array to load following the first eclipse. All signals were nominal until 11-21-68, then ceased. Returned to functional status 2-24-69. No intermittency noted during Fall 1969 or Spring 1970 eclipse seasons. All signals ceased 12-17-70. Recommenced 2-1-71. All signals ceased 12-1-71; recommenced 2-16-72. Switched during lunar eclipse; suspected cause is RCU failure. All signals ceased 4-22-73; recommenced 8-17-73. All signals ceased 5-9-74; recommenced 5-22-74. |
| 9345 | Timed out | TLM not operating at 1315Z and 2230Z on 1-13-70. Operation regained 2-24-70; ceased 10-19-74. | Timed out | E<br>Switched: 2-18-69 | WJ | Switched following lunar eclipse; suspected cause is TWTA degradation and RCU failure. Communications not operating at 0200Z 1-13-70. Returned to functional status 2-24-70; ceased 10-19-74, probable time-out. |

*History of intermittent performance.

Table C.6-5. On-Orbit Malfunction Summary – Satellites 9341 through 9348, Launched June 1968 (Cont.)

| | Telemetry Characteristics | | Communications Characteristics | | | |
|---|---|---|---|---|---|---|
| | | | | TWTA | | |
| Sat. ID. | Performance | Comments | Performance | I | R | Comments |
| 9346 | Ceased | Telemetry ceased as of 9-23-68. | Nominal* | WJ Switched: 3-30-68 | E | Switching caused by TWTA failure. TWTA operation degraded continuously since launch. All signals ceased 9-23-68 following an eclipse. Communications capability returned 9-24-68. Possible intermittent PCU. All signals ceased 12-1-71; recommenced 5-5-72. All signals ceased 10-19-72. Recommenced 8-27-73. |
| 9347 | Ceased* | Inoperative 6-1-69 to 8-28-69 Ceased 11-26-73. | Ceased* | E Switched: 6-14-68 | WJ | Suspected switching cause is failed RCU after 22 hours in orbit. All signals ceased 6-1-69, recommenced 8-28-69. Classified as a failed unit 12-1-71 due to low ERP. Ceased 11-26-73. |
| 9348 | Timed Out | Telemetry intermittent subsequent to Spring 1970 eclipse. Telemetry ceased 9-17-70. Possibly related to communications failure. Operation regained 2-22-71. All telemetry ceased as of 12-1-71, recommenced 2-21-72. Ceased 11-19-73; recommenced 2-13-74. All telemetry ceased 10-21-74. | Timed Out | WJ Switched: 11-5-71 | E | All signals ceased 9-17-70. Prior to failure, health was nominal. Cause of failure is unknown. Returned to functional status 2-22-71. Suspected switching cause is unknown. All signals ceased 12-1-71; recommenced 2-21-72. Classified as a failed unit 11-20-72 due to suppressed ERP. Reclassified as operational 2-12-73. Ceased 11-19-73; recommenced 2-13-74. All signals ceased 10-21-74. Cause is RCA/PCU or time-out. |

*History of intermittent performance.

References

C.6-1. Orbital Performance Report, DSCP Satellites, TOR-0075(5403-01)-5, The Aerospace Corporation, El Segundo, California (31 December 1974).

C.6-2. Statistical Analysis of TWTA Failure Timer, TOR-0073(3701)-3, The Aerospace Corporation, El Segundo, California (30 August 1972).

## C.7 TACTICAL COMMUNICATIONS SATELLITE (TACSAT)

### C.7.1 Program Summary

TACSAT was designed and fabricated by the Space and Communications Group, Hughes Aircraft Company, at its El Segundo, California facility. This activity was under the direction of the Air Force Space and Missile Systems Organization (SAMSO). The Aerospace Corporation was the General Systems Engineering/Technical Direction contractor.

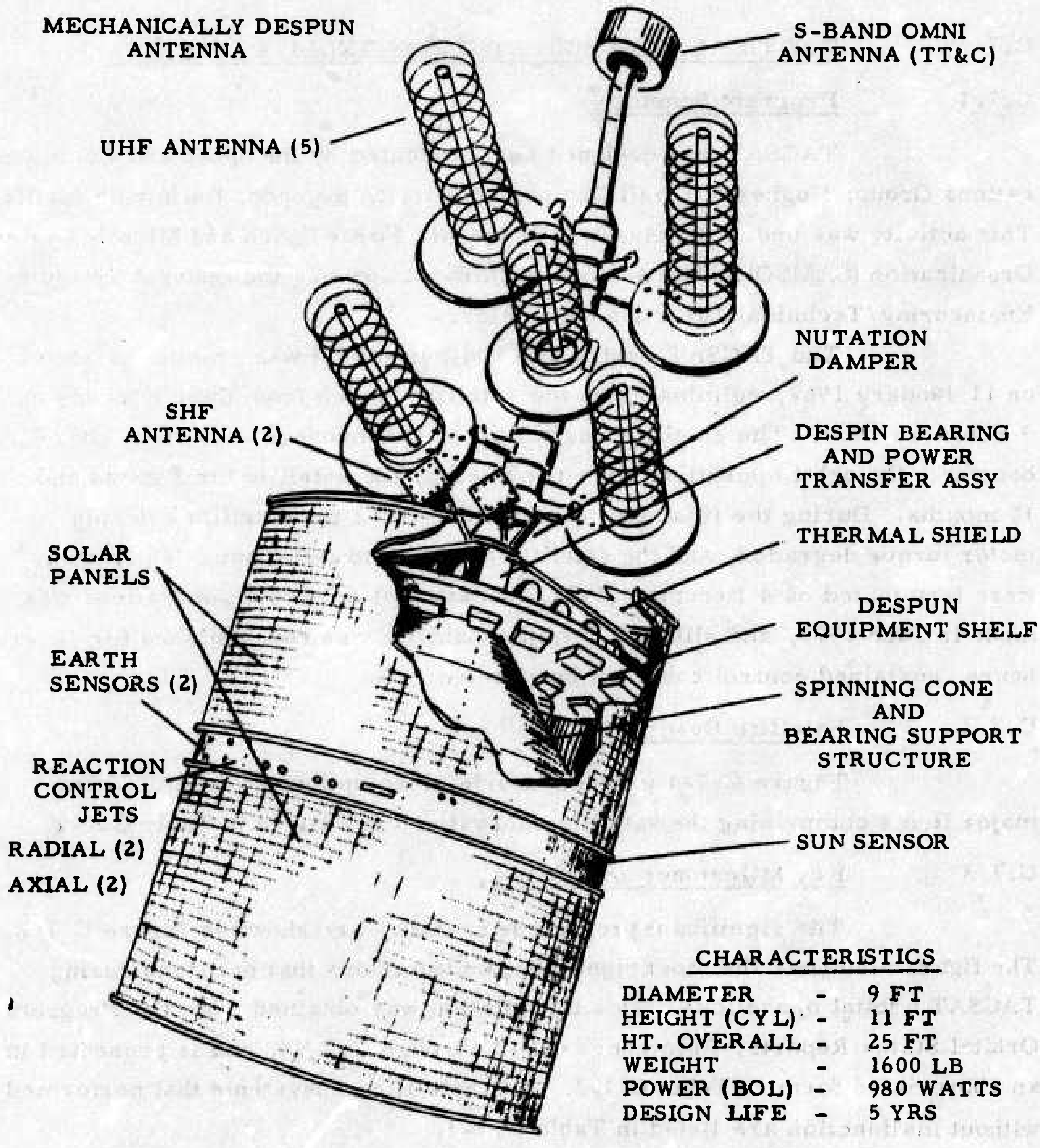
The TACSAT contract [AF04(695)-1047] was granted go-ahead on 11 January 1967, culminating in the satellite launch from Cape Kennedy on 9 February 1969. The satellite was placed in synchronous orbit by a Titan IIIC booster. On-orbit operations were provided by the satellite for 3 years and 10 months. During the first days of December 1972 the satellite's despin motor torque degraded, and the satellite settled into a flat spin. Operations were terminated on 4 December 1972. An attempt to recover operations was made in June 1973, and although despin capability was reestablished for several hours, sustained control could not be effected.

### C.7.2 Satellite Description

Figure C.7-1 presents a brief description of TACSAT. The major items comprising the satellite subsystems are listed in Table C.7-1.

### C.7.3 Key Milestones and Events

The significant program milestones are shown in Figure C.7-2. The figure also lists the most significant malfunctions that occurred during TACSAT orbital operations. This information was obtained from the Program Orbital Status Reports, References C.7-1 through C.7-19, and is presented in an abbreviated form in Table C 7-2. The satellite subsystems that performed without malfunction are listed in Table C.7-3.




**CHARACTERISTICS**

| | | |
|---|---|---|
| DIAMETER | - | 9 FT |
| HEIGHT (CYL) | - | 11 FT |
| HT. OVERALL | - | 25 FT |
| WEIGHT | - | 1600 LB |
| POWER (BOL) | - | 980 WATTS |
| DESIGN LIFE | - | 5 YRS |

| | | |
|---|---|---|
| LAUNCHED | - | 9 FEB. 1969 |
| BOOSTER | - | T-III C |
| ORBIT | - | SYNC. EQUATORIAL |

Figure C.7-1. Tactical Communications Satellite (TACSAT)

## Table C.7-1. TACSAT Subsystem Description

<u>Structure</u>

- Cylindrical solar panel substructures (2)
- Spinning cone inner structure with spin bearing support (54 rpm)
- Despun equipment platform and antennas

<u>Orbit and Attitude Control</u>

- Reaction control
  - $H_2O_2$ propellant tanks (4)
  - Axial jets (2), radial jets (2)
  - Jet control electronics (JCE)
- Attitude determination
  - Earth sensors (2), sun sensor (all sensors with dual heads)
  - Ground computation of attitude

<u>Despin/Dynamics</u>

- Despin control electronics (DECEL) redundant units
- Bearing and power transfer assembly (BAPTA)
  - Despin torque motor
- Nutation damper

<u>Electrical Power</u>

- Solar arrays - forward and aft cylindrical panels (spun)
- Batteries - NiCd (3), each 6 A-hr
- Power conditioning - battery controllers (3)
  - Voltage limiters (10)
  - Current sensors (5)
- Distribution - primary and secondary buses

<u>Telemetry, Tracking, and Command</u>

- S-band telemetry and command - SGLS compatible
  - Despun omni antenna
  - Redundant command receivers plus 4 decoders (cross-strapped)
  - Redundant TLM encoders (4) - automatic switching

## Table C.7-1. TACSAT Subsystem Description (Cont.)

Thermal Control

- Passive thermal control
  - Forward sunshield
  - Aft thermal barrier
  - Thermal design for equipment power dissipation

Repeater Subsystem

- Multiple channels, 50 kHz to 10 MHz bandwidths
- Capacity - nominally equivalent to 20,000 telephone circuits with standard Intelsat ground stations.
  - UHF - About 40 vocoded voice or several hundred teletype circuits to a terminal with 0 dB antenna gain
  - SHF - About 40 vocoded voice or 700 teletype circuits to a terminal with 3 ft, antenna
- Antenna
  - UHF - 5 bifilar helices, 17.1 dB transmit gain peak, 17.6 dB receive gain peak
  - SHF - Transmit: horn, 18.4 dB gain peak, 19° beamwidth
    - Receive: horn, 19.3 dB gain peak, 17° beamwidth
- Transmitters - 249.6 and 7257.5 MHz
  - UHF - Solid state, 16 parallel amplifiers, nominal 13 on
  - SHF - 3 TWTs, 2 on at a time
- Receivers - 303.4, 307.5, and 7982.5 MHz
  - UHF - Transistor amplifiers, 3.7 dB noise figure
  - SHF - Tunnel diode preamp, 6.9 dB noise figure

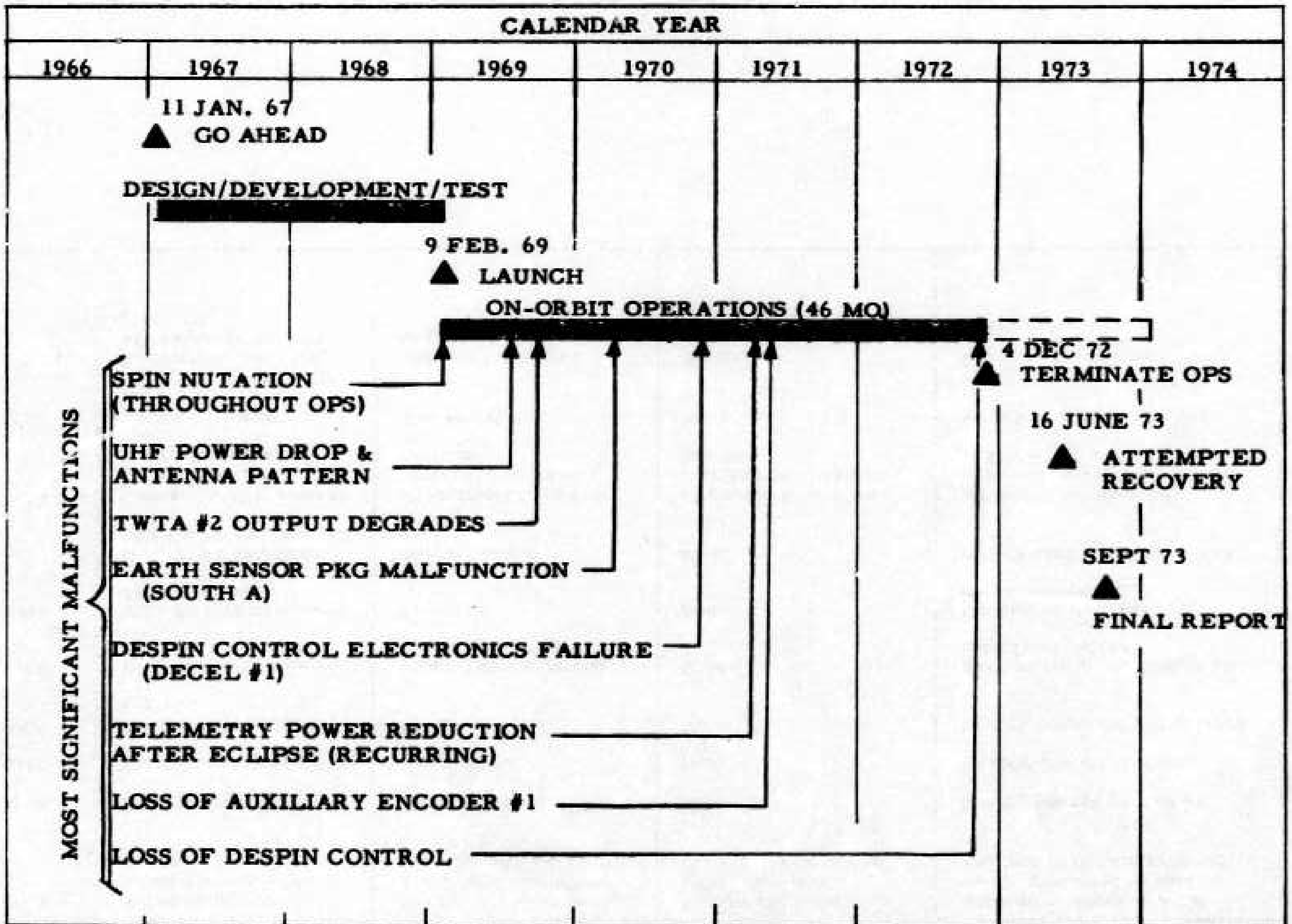



Figure C.7-2. Key Milestones and Events — TACSAT

Table C.7-2. On-Orbit Malfunction Summary - TACSAT, Launched 2/9/69

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 2/69 | Nutation approx. 1.1° failed to damp after spinup | Energy dissipation de-damping due to oil in rotating BAPTA shaft clearances | Persisted - small effect on comm. system performance | Attempt to damp in ACM mode; continued to analyze and test to isolate problem |
| 3/69 | Battery overcharge | Cold batteries | None | Modify charge procedure |
| 3/69 | Earth sensor anomalies | Sun viewing | None | Review ground test data |
| 3/69 | Bit shift in command register | Unknown | None | Modify command procedures |
| 3/69 | RCS tank temperature anomalies | Suspect temp. transducer unbonding | Minor | Rely on temp. indication and calculated values |
| 4/69 | UHF ERP level erratic during eclipse | Unknown | None | Ground station test/measurements |
| 6/69 | DECEL #1 switch to DECEL #2 automatically | Suspect noise | None | Switch back to primary unit |
| 7/69 | Change in UHF antenna pattern and 3 to 5 dB drop in power level | Suspect collison with foreign object and antenna | Degraded UHF performance throughout mission | Switch off temporarily, simulate by test and analysis |
| 9/69 | TWTA #2 output degraded | Power supply | None | Switch to redundant units |
| 9/69 | Inadvertent switching of equipment on despun platform | Suspect auxiliary decoder #1 | None | Reset by commands |

Table C.7-2. On-Orbit Malfunction Summary - TACSAT, Launched 2/9/69 (Cont.)

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 10/69 to 1/70 | Improper maneuvers for attitude correction | Incorrect procedures commanded erroneous jet firings | Two extra maneuvers required | Revised procedures |
| 4/70 | Earth sensor pkg. failed (South A) | Not known | None | Switch to redundant units |
| 3/70 to 4/70 | Earth sensor anomalies (North B) | Sun blinding | None | Switch to redundant units |
| 10/69 to 11/70 | Automatic switching of DECEL #1 to DECEL #2 (numerous instances) | Suspect noise pulse or sun interference | None | Command switch back to DECEL #1 |
| 11/70 | DECEL #1 failure | Phase comparator not processing earth pulses | Minor | Switch to redundant DECEL #2 |
| 11/70 | Platform spinup to 10 rpm inadvertently | Unplanned during diagnostic test of DECEL #1 | Suspect additional damage to UHF antenna | Despin platform |
| 11/70 | Decrease in UHF power level | Antenna change during spinup | 2 dB drop in power | Attempt pointing ofsets |
| 12/70 | Battery #1 voltage drop | All cells not discharging | None | Monitor and implement recharge procedures |
| 3/71 | DECEL #2 gain changes | Spurious signals by aux. decoder #2 | Operational complexity | Reset DECEL gain by command |
| 3/71 | T/M power reduction | Suspect intermittant connection in multiplexer | Loss of low angle station contacts ($K_oD_1$) | Attempt operational work-arounds |

Table C.7-2. On-Orbit Malfunction Summary - TACSAT, Launched 2/9/69 (Cont.)

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 4/71 to 9/72 | Temporary losses of despin control (numerous instances) | Suspect earth sensor malfunctions | Antenna pointing error | None - DECEL #1 had previous failure 11/70 |
| 5/71 to 9/72 | Uncommanded status changes for equipment on despun platform | Undertermined, related to turn-on of aux. decoder #1 | Operational complexity | Reconfigure by commands at station contacts |
| 5/71 | Aux. encoder #1 malfunction | Data node switch suspected | Loss of some temp. data | Access for brief period acquired good data |
| 12/71 to 9/72 | Command system rejected commands (several instances) | Commands converter #1 out of synch. with ground station | Loss of commands | SCF perform echo checks and switch to redundant system |
| 12/72 | Loss of despin control | Loss of motor torque | Flat spin and loss of antenna pointing | Terminate on-orbit operations |

Table C.7-3. Summary of Equipment Performing Without Malfunction In-Orbit

Reaction Control Subsystem

With the exception of tank temperature transducers that apparently became unbonded, the RCS subsystem performed as required during the life of the satellite.

Thermal Control Subsystem

Equipment temperatures were maintained within temperature limits during the life of the satellite following nominal expected values for conditions of spacecraft power, repeater configuration, and sun vector angle.

Telemetry, Tracking, and Command

With the exception of auxiliary encoders, decoders, receiver and transmitter performed well. A drop in telemetry power approximately halfway through the mission was attributed to a faulty RF multiplexer.

Electrical Power

Solar panel performance after 42 months in orbit correlated well with calculated predictions; output degradation including effect of the August 1972 solar flare was approximately 6 percent of the usable power. The battery controllers, voltage limiters, and current sensors operated satisfactorily over the life of the satellite.

Malfunctions That Corrected Themselves

The following items of equipment apparently recovered in some manner from a prior condition of failure or degraded performance:

- Earth sensor (South A) - failed 4/70 - appeared normal on 3/71.
- Battery #1 - improper behavior 12/70 - acceptable after 3/71.
- DECEL #2 - intermittent control loss stopped from 2/72 to 8/72.
- TLM signal strength dropped after eclipses apparently due to temperature sensitive RF multiplexer - recovered after eclipses.

## C.7.4 Satellite Orbital Experience

From some 25 itemized malfunctions given in Table C.7-2 the majority were categorized as anomalies with little or no impact on the mission. However, eight events were identified as significant malfunctions in terms of failed equipment or events having a deleterious effect on mission performance.

As shown on Figure C.7-2, three failures concerned the Despin/ Dynamics Subsystem, three were in the Telemetry and Command Subsystem, and two were related to the Communications Repeater Subsystem.

The history of TACSAT was evaluated by the SAMSO Program Review Board in 1972 and the findings are presented in References C.7-20 through C.7-22. A brief recapitulation of those findings is given below.

### C.7.4.1 Mechanical Malfunctions

The TACSAT nutation was attributed to the Attitude Control Subsystem at the component level and involved the dynamics of the despin bearing. The Design, Test and Human/Procedural Working Group concluded that the problem was very subtle, encompassing both the design concept and its implementation. Tests adequate to explore the problem could have provided prior knowledge of the design interactions, but it was considered that, because of the complexity involved, it was unlikely the problem could have been anticipated.

The group's comments on deployable devices stated that force and torque margin requirements are related to the sufficiency of the analysis in properly considering all of the factors involved in the determination of both design loads and driving capability. It is not always possible to get valid test verification because of the difficulty in obtaining a representative simulation of the actual operating conditions. Since large margins for these devices are often obtainable with little or no system penalty, the criterion should not be standardized to a constant factor, such as is done for structural margins. Rather, margins should be established for each device based on a tradeoff study which considers design uncertainties, risk, and system penalties. It was recommended that the statement of work in future contracts should include the following requirements for mechanical or deployable devices:

a. Tradeoff to establish penalties with increasing force/torque margins

b. Analysis for both load and driving capability based on worst case combined conditions

c. Design consideration given to anomalies/failure conditions

d. Specified margin demonstrated by test, based on worst-case flight conditions

Specific design recommendations from the TACSAT Program Office applicable to Intelsat IV were to redesign the bearing assembly to eliminate viscous damping on the spinning side of the interface and install a more efficient nutation damper on the platform.

The UHF antenna problem was investigated by means of analyses, breadboard tests, on-orbit measurements, and scale model measurements. No failure mode due to design deficiency was identified and it was suspected that damage to the antenna may have been caused by a collision with space debris such as a micrometeorite.

C.7.4.2 Electronic Malfunctions

In the cases of electronic equipment malfunctions, the Quality Assurance Working Group of the Program Review Board performed an audit of the contractor's facility and concluded that workmanship problems could have contributed to the on-orbit failures. It was recommended that hardware audits be used by the System Program Office and the contractor to evaluate the adequacy of workmanship, process control, and package design. A brief summary of the findings of that group is as follows:

a. Despin Electronics Control (DECEL) - The DECEL #1 failure was considered to be a thermally induced problem in the electronic shaft angle encoder (ESAE) causing the circuitry to saturate at its upper level. A large variety of open or short type component or interconnection failures in the ESAE could cause this saturation. The potential problems were isolated to the digital and analog A3 and A4 boards, respectively.

Time did not permit an in-depth analysis of contractor build, test, and failure data, but review of contractor reports indicated that the contractor had accomplished an in-depth analysis. The report and discussions with the contractor indicated that the boards had been subjected to cleaning operations that had not been performed satisfactorily, resulting in damage to components mounted on the printed circuit boards and broken or bent leads. Some of these boards, which were considered substandard, were flown. The contractor did not believe that this was contributory to the on-orbit problem.

A spare DECEL unit was provided and disassembled and inspected by the subgroup team. In particular, boards A3, A7, and A16 were examined. Workmanship problems relating to lack of stress relief of component leads and variation in amount of tinning on leads was evident. In some cases, gold-plated leads appeared to have no tinning. Some integrated circuits had tinning on the assembly leads that touched the case. Other portions of this assembly had dense packaging arrangements that made inspection difficult and unsatisfactorily routed wires that could cause shorts due to cold flow of insulation. Time did not permit examination of records to determine adequancy of manufacturing/inspection planning, but the discrepancies cited above suggest lack of control during the lead tinning operation.

b. TWTA Anomaly - A review of the telemetry data between March and September 1969 indicates a gradual decline in SHF power monitor reading of about 0.5 dB. Whether this was an actual decrease or long-term drift in the power monitor/telemetry system is somewhat uncertain. However, measurements of spacecraft ERP at Camp Parks are consistent with an ERP decrease of the same magnitude, although a data scatter of about 1 dB obscures that conclusion.

The apparent problem with TWTA 2 prompted a review of acceptance test data on that component. The only anomaly is a tendency to higher-than-normal helix current at high temperature, e.g., 140°F. This condition as tested would not affect operation. However, if the condition worsened with time or became more temperature sensitive, the increased helix current could interact with the power supply to change cathode voltages and, hence, the phase shift characteristic of the tube. To produce a loss of power of 1.5 dB over nominal, a voltage change of about 40 V would be required.

A review of other possibilities indicates that phase shift is the most likely explanation of the problem; however, a definitive solution does not appear possible with the available data. Discussion with the Hughes engineering personnel indicates the phase shift problem is most likely caused by a part failure in the power supply.

MIL-Q-9858A was the basis for the quality requirements for the contract, AF04(695)-1047. Workmanship standards were in accordance with Hughes Aerospace Group standards. A contractual quality assurance plan CPD 28721400 was on contract and approved by the SPO. ECP 013 dated 10 May 1969 changed the plan to conform to Hughes established procedures and was revised and approved by the SPO. The quality plan was satisfactory for the implementation of MIL-Q-9858A.

The findings of the group were that inadequate workmanship existed on the printed circuit assemblies and that substandard material was used for flight. It was concluded that the workmanship problems could have contributed to the on-orbit failure of the DECEL, that the contractor's inspection was inadequate, and that process control for lead tinning also was inadequate.

The particular events of ground test malfunctions given in Table C.7-4 do not appear to relate directly to the on-orbit malfunctions previously discussed. However, the consequences of not detecting the faults cited would have had a major impact on the mission in one case, and a somewhat less deleterious effect in the other two cases. There may have been some correlation between the instance of tightened wireclamps and the foregoing discussion of detail procedures at the launch base. In that particular case, the value of a final verification test after mating to the booster was demonstrated.

Table C.7-4. Malfunctions During Pre-Launch Ground Tests (Eastern Test Range) - TACSAT

| Malfunction | Response | Consequence If Undetected |
|---|---|---|
| Solar Panel Output Anomalies | Contractor replaced defective solar cell at launch base | Low power at end of mission life |
| Voltage B+ Shorted to Ground | A technician tightened several wire clamps after the final system test at SAB (no procedure, no inspection). This malfunction would have remained undetected had the satellite not been tested on the launch vehicle | Catastrophic failure and loss of mission |
| Failure of Command System on The Launch Stand | Unit replaced | Reduction in reliability of command control to one half |

## References

Except where noted, all references were generated by Hughes Aircraft Company, Space Systems Division.


C.7-1. MILCOMSAT - Ortibal Operations Report No. 1, Contract No. F04701-69-C-0075 (March 1969).

C.7-2. Orbital Status Report, Contract No. F04701-69-C-0079 (April 1969).

C.7-3. TACSAT - Orbital Operations Report No. 2, Contract No. F04701-69-C-0079 (May 1969).

C.7-4. TACSAT - Orbital Status Report No. 2, Contract No. F04701-69-C-0079 (June 1969).

C.7-5. TACSAT - Orbital Status Report No. 3, Contract No. F04701-69-0079 (July 1969).

C.7-6. TACSAT - Orbital Operations Report No. 3, Contract No. F04701-69-C-0079 (October 1969).

C.7-7. TACSAT - Orbital Operations Report No. 4, Contract No. F04701-69-C-0079 (February 1970).

C.7-8. TACSAT - Orbital Status Report, Contract No. F04701-70-C-0089 (May 1970)

C.7-9. TACSAT - Orbital Status Report, Contract No. F04701-70-C-0089 (September 1970).

C.7-10. TACSAT - Orbital Status Report, Contract No. F04701-70-C-0089 (November 1970).

C.7-11. TACSAT - Orbital Status Report, Contract No. F04701-70-C-0089 (January 1971).

C.7-12. TACSAT - Orbital Status Report, Contract No. F04701-70-C-0089 (April 1971).

C.7-13. TACSAT - Orbital Status Report, Contract No. F04701-70-C-0089 (June 1971).

C.7-14. TACSAT - Orbital Status Report, Contract No. F04701-71-C-0302 (October 1971).

C.7-15. <u>TACSAT - Orbital Status Report</u>, Contract No. F04701-71-C-0302 (January 1972).

C.7-16. <u>TACSAT - Orbital Status Report</u>, Contract No. F04701-71-C-0302 (June 1972).

C.7-17 <u>TACSAT - Orbital Status Report</u>, Contract No. F04701-72-C-0319 (September 1972).

C.7-18. <u>TACSAT - Orbital Status Report</u>, Contract No. F04701-72-C-0319 (January 1973).

C.7-19. <u>TACSAT - Final Report</u>, Contract No. F04701-72-C-0319 (September 1973).

C.7-20. <u>Quality Assurance Working Group Report</u>, U.S. Air Force Space and Missile Systems Organization, El Segundo, California (August 1972).

C.7-21. <u>Electronic Piece Parts Working Group Report</u>, TR-0073(3921-02)-1, The Aerospace Corporation, El Segundo, California (15 January 1973).

C.7-22. <u>Design, Test and Human/Procedural Working Group Report</u>, TOR-0073(3921-01)-1, The Aerospace Corporation, El Segundo, California (4 December 1972).

## C.8 SKYNET I/NATO II COMMUNICATIONS SATELLITES

### C.8.1 Program Summary

The Skynet I and NATO II spacecraft were designed and fabricated by the Space and Re-Entry Systems Division of the Philco-Ford Corporation of Palo Alto, California. The program was under the direction of the Air Force Space and Missile Systems Organization (SAMSO) with General Systems Engineering/Technical Direction performed by The Aerospace Corporation.

The Skynet I contract was granted go-ahead in March 1967, culminating in the launch of the first of two satellites on a Thor-Delta booster on 24 November 1969 and the second on 19 September 1970. The Thor-Delta placed the satellites in 100 by 19,300 nmi transfer orbits with synchronous equatorial orbit attained by the satellite apogee boost motor. The first satellite (SK-IA) operated until November of 1971 when the second, of two, TWTAs failed and the spacecraft could no longer perform its communications mission. The second satellite (SK-IB) was lost during burn of the apogee boost motor in the transition from the transfer orbit to synchronous orbit.

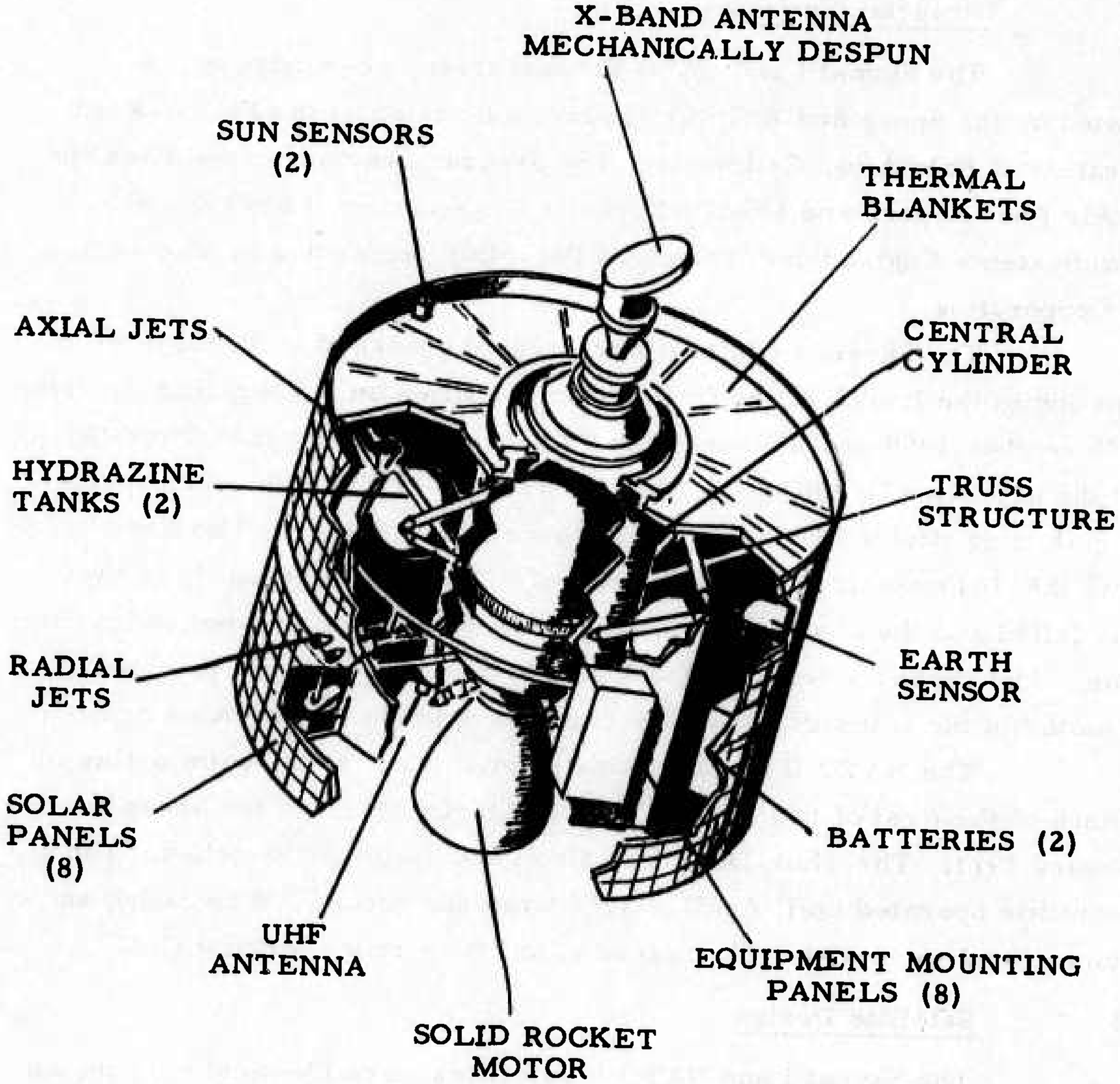
The NATO II contract started on 3 April 1968, culminating in the launch of the first of two satellites on 20 March 1970 and the second on 2 February 1971. The Thor-Delta was also used for these launches. The first satellite operated until April of 1972 when the second TWTA failed and communications were lost. The second satellite is still operational.

### C.8.2 Satellite Design

The Skynet I and NATO II satellites were identical with the exception of slight differences in the X-band antenna beam shape and the power split between the 2 and 20 MHz channels. The satellites are depicted in Figures C.8-1 and C.8-2. The major items comprising the satellite subsystems are listed in Table C.8-1.

### C.8.3 Key Events and Milestones

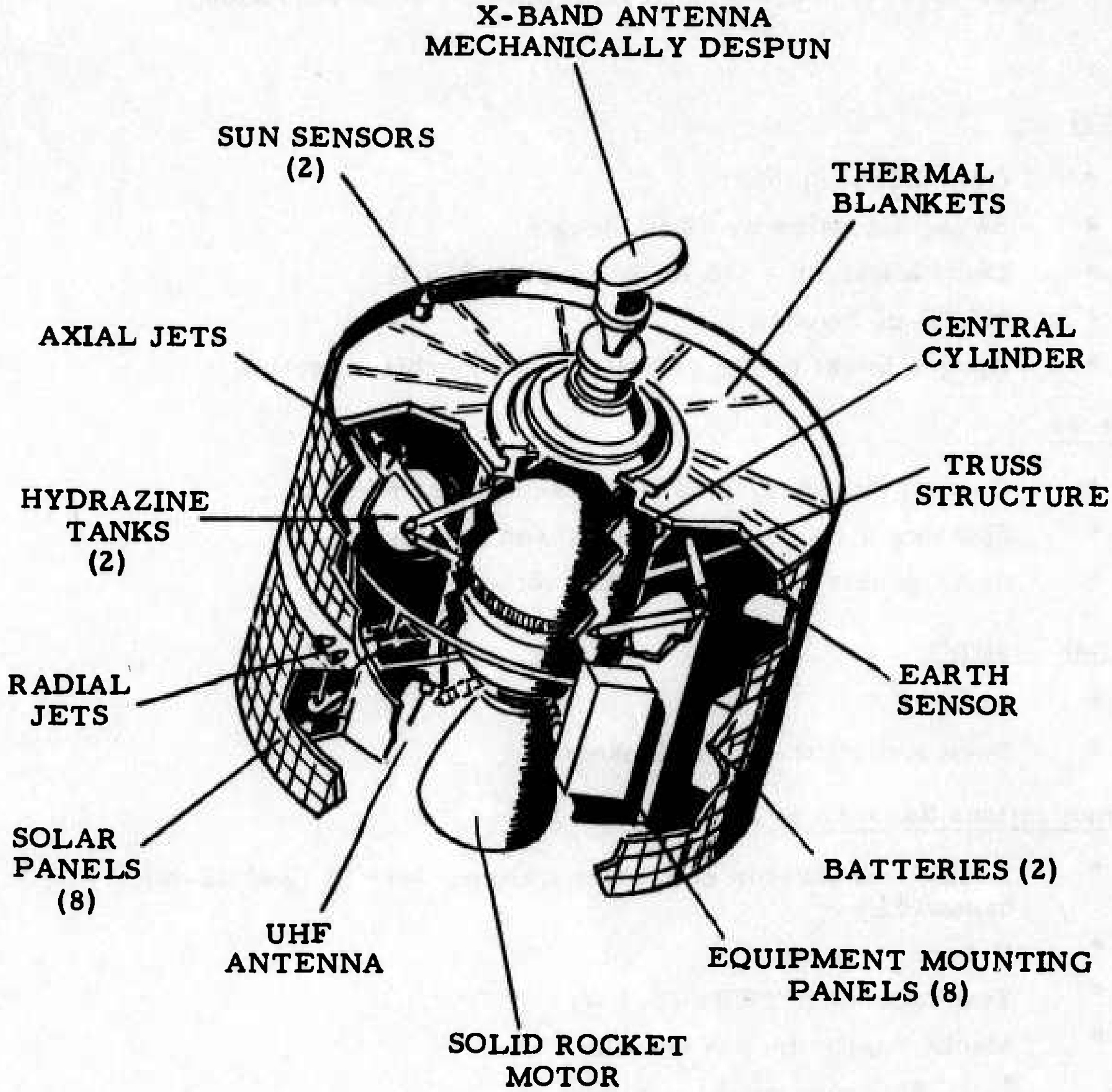
The significant program milestones are shown in Figures C.8-3 and C.8-4, which also list the most significant malfunctions that occurred




| LAUNCHES | |
|---|---|
| SK-1A | - 21 NOV 1969 |
| SK-1B | - 19 AUG 1970 |
| BOOSTER | - THOR DELTA |
| ORBIT | - SYNC. EQUATORIAL |

| CHARACTERISTICS | |
|---|---|
| DIAMETER | - 54 IN |
| HT. (OVERALL) | - 61.8 IN |
| WT. ( LIFT OFF) | - 535 LB |
| POWER (BOL) | - 113 WATTS |
| DESIGN LIFE | - 3 YEARS |

Figure C.8-1. Skynet I Communications Satellite




| LAUNCHES | | |
|---|---|---|
| NATO IIA | - | 20 MAR 1970 |
| NATO IIB | - | 3 FEB 1971 |
| BOOSTER | - | THOR DELTA |
| ORBIT | - | SYNC. EQUATORIAL |

**CHARACTERISTICS**

| | | |
|---|---|---|
| DIAMETER | - | 54 IN |
| HT. (OVERALL) | - | 61.8 IN |
| WT. (LIFTOFF) | - | 535 LB |
| POWER (BOL) | - | 113 WATTS |
| DESIGN LIFE | - | 3 YEARS |

Figure C.8-2. NATO II Communications Satellite

Table C.8-1. Skynet I/NATO II Subsystem Description

<u>General</u>

- Cylindrical spinner
- 54 in. diameter by 62 in. length
- Launch weight - 535 lb, on-orbit - 285 lb
- MTTF of 3 years
- Apogee boost motor (ABM) for final orbit insertion

<u>Structure</u>

- Central cylinder, truss, and equipment panels
- Spinning structure/despun X-band antenna
- Solar panels mounted on structure

<u>Thermal Control</u>

- Passive
- Fore and aft thermal blankets

<u>Communications Subsystem</u>

- Double-conversion redundant transponders (2- and 20-MHz bandwidth)
- X-band frequency
- Two redundant TWTs (3.5 W)
- Mechanically despun antenna
  - Skynet - earth coverage
  - NATO - Northern Hemisphere from U.S. to Turkey
- X-band beacon

<u>Apogee Boost Motor</u>

- Solid propellant
- Thrust - 4500 lb
- Fired by ground command

Table C.8-1. Skynet I/NATO II Subsystem Description (Cont.)

<u>Orbit and Attitude Control</u>

- Reaction Control
  - $N_2H_4$ blowdown propellant tanks (2) and plumbing
  - Axial jets (2), radial jets (2), 5 lb thrust each
  - Control electronics
- Attitude Determination
  - Earth sensors (2) with sun guard sensors
  - Sun sensors (2)
  - Sun angle sensor
- Nutation Damper
  - Ball in tube
- Ground computation of attitude
- Control electronics for X-band antenna

<u>Electrical Power</u>

- Body-mounted solar arrays
- Two batteries (NiCd)
- Solar array/battery controls
- Distribution and power control circuitry
- Redundant UHF antenna arrays

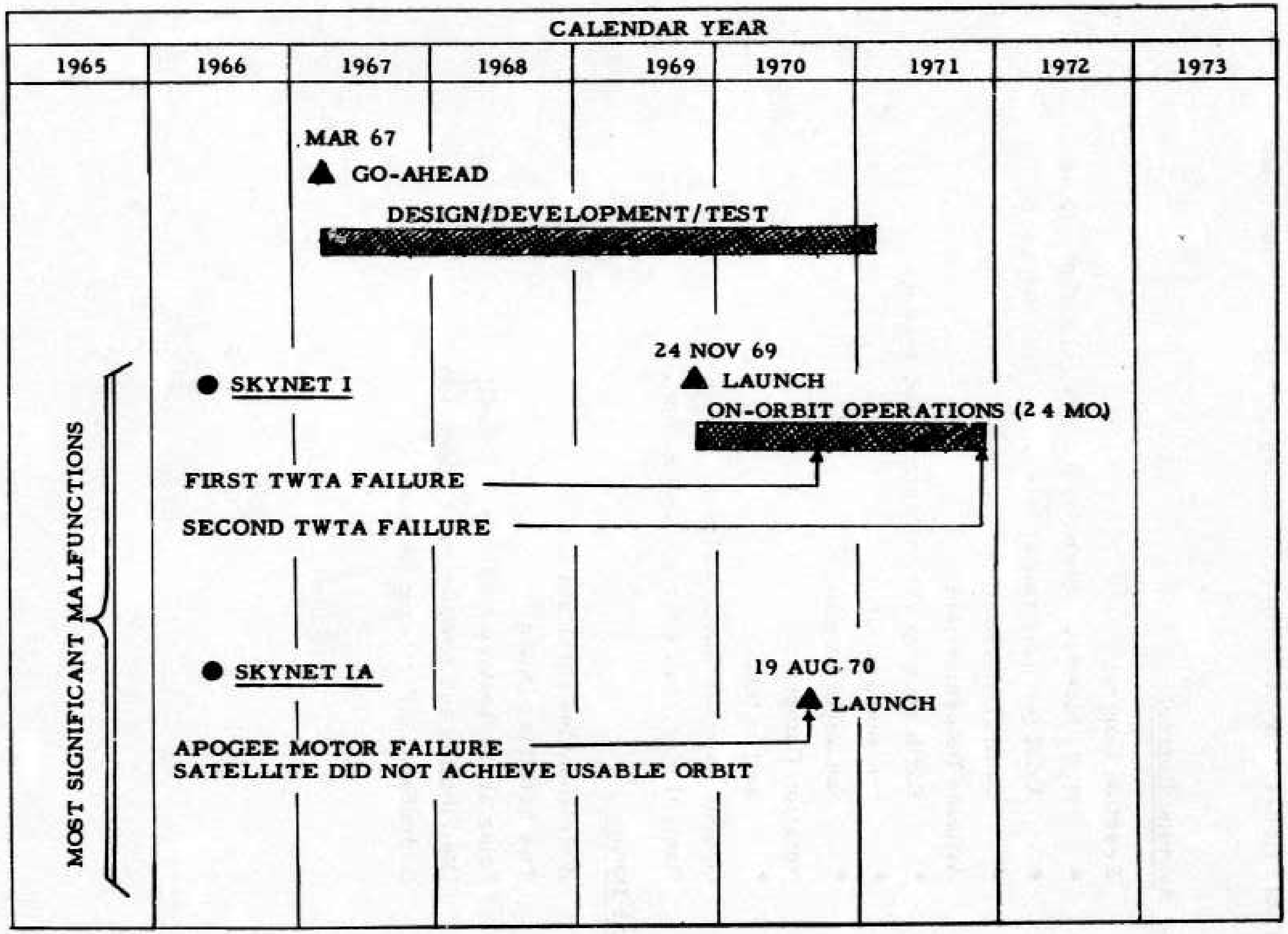



Figure C. 8-3. Key Milestones and Events - Skynet I

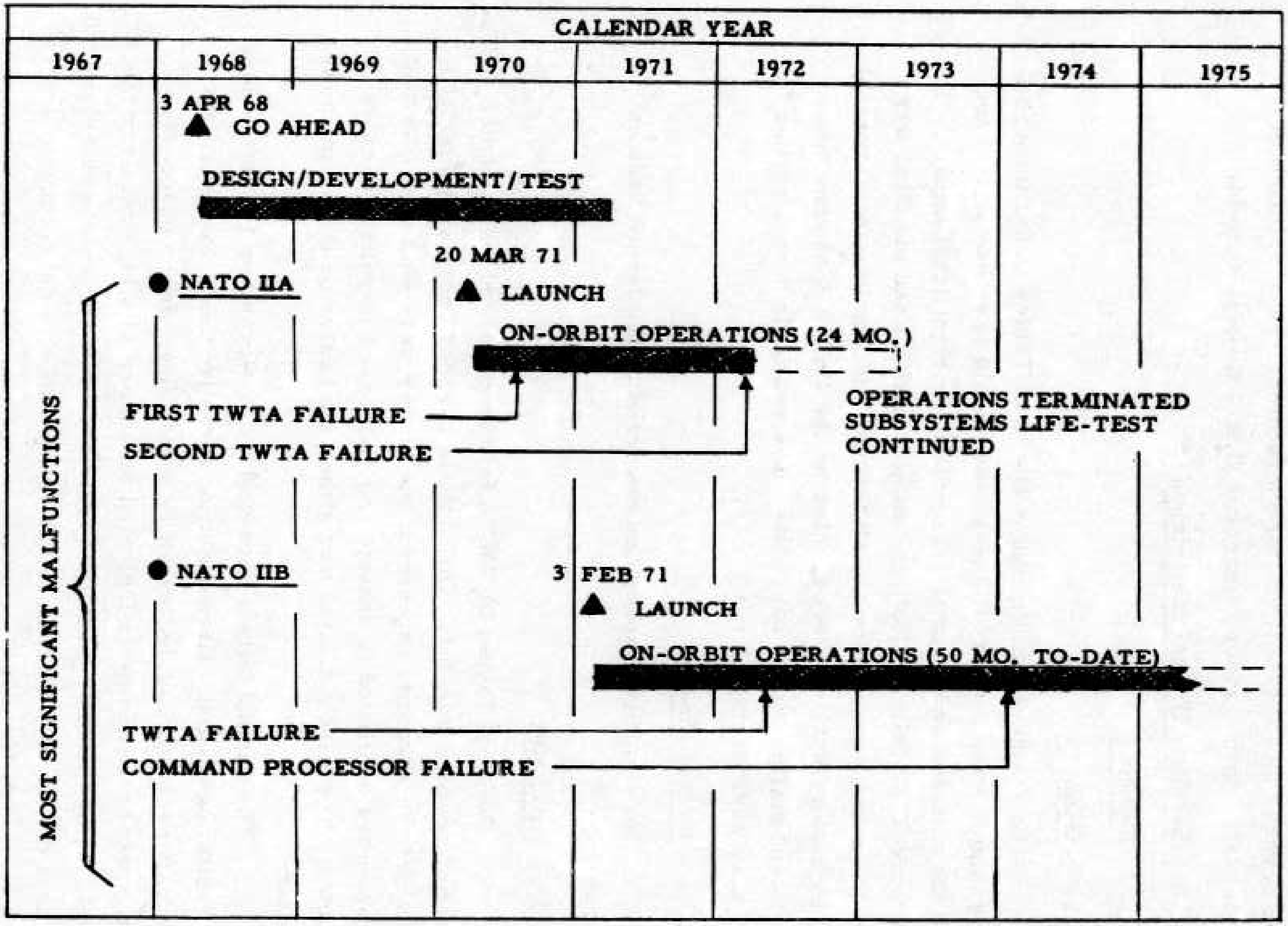



Figure C. 8-4. Key Milestones and Events - NATO II during the Skynet I and NATO II orbital operations. This malfunction information was obtained from References C.8-1 through C.8-14.

## C.8.4 Satellite Orbital Experience

### C.8.4.1 Skynet I

#### C.8.4.1.1 General

Of the seven malfunctions listed in Tables C.8-2 through C.8-5, one during launch base testing, if undetected, could have had significant impact on the mission and three in space did have significant impact. Two of the latter were TWT failures (both the same satellite) and the third was a catastrophic apogee boost motor (ABM) failure. (Since long-term control of Skynet IA in the final orbit was handled by the United Kingdom, other less significant malfunctions, not contained in available documentation in the U.S., may have occurred.)

#### C.8.4.1.2 Launch Base

No significant anomalies occurred at the launch base on Skynet I spacecraft.

#### C.8.4.1.3 On-Orbit

The first Skynet IA TWTA failure occurred on 8 October 1970 in a matter of several minutes. The cathode current decay characteristic was similar to the response expected if the heater were shut off. The high-voltage converter continued to operate and increased in voltage because it was unloaded. The data indicated that either the heater or the heater power supply failed.

The second failure occurred on 29 November 1971 after 9480 hours of orbital operation. Ground station measurements confirmed that the spacecraft receiver had decreased in sensivitity. In the unaccessed condition, spacecraft telemetry indicated an abnormally high receiver input noise level. It was further confirmed from ERP measurements at the ground terminal that the useful signal from the spacecraft was decreasing at a

Table C.8-2. Launch Base Malfunction Summary - Skynet IA

| Malfunction | Cause | Potential Impact on Mission | Corrective Action |
|---|---|---|---|
| Battery appeared shorted | AGE wiring | Reduced eclipse capability | Replaced battery |
| Battery connector broken | Improper installation | Minor | Epoxied in place |

C.8-3. Launch Base Malfunction Summary - Skynet IB

| Malfunction | Cause | Potential Impact on Mission | Corrective Action |
|---|---|---|---|
| Communications subsystem spurious signals. | Design limitation | Minor | Waiver |
| ABM alignment out-of-limit | Tolerance buildup | Minor | Waiver |

Table C. 8-4. On-Orbit Malfunction Summary - Skynet IA

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 10/70 | TWT failed | TWT heater | Loss of redundancy | Switched to redundant tube |
| 11/71 | TWT failed | Out-of-band oscillation | Catastrophic | None |

Table C. 8-5. On-Orbit Malfunction Summary - Skynet IB

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 2/71 | ABM failed during firing | Moisture in propellant | Catastrophic | None |

greater rate than that seen on RF output power telemetry. No discrete spurious signals were evident. It was concluded that the TWTA had become unstable and was oscillating at a frequency outside the communications frequency band.

All subsystems of the Skynet IB spacecraft were operating nominally when the ABM ignition command was transmitted to the spacecraft. Ignition was confirmed and measurements on telemetry were nominal until 13.9 sec after ignition (the nominal burn was 22 sec) when telemetry abruptly ceased. Further contact with the spacecraft was never re-established. The Skynet IB Failure Review Board established (Ref. C.8-2) that "the quality assurance function in the ABM manufacturing process needs attention." This was based on the conclusion that inadequate humidity control during casting of the motor could have resulted in excessive moisture in the propellant and inadequate propellant bonding.

### C.8.4.2 NATO II

#### C.8.4.2.1 General

From the 26 malfunctions given in Tables C.8-6 through C.8-9, one during launch base testing, if undetected, could have had a significant impact on the mission and four in space did have significant impacts. Three of the latter were TWT failures (the two on NATO IIA terminated the satellite mission) and the fourth resulted in loss of one of the two command units.

#### C.8.4.2.2 Launch Base

A capacitor with the wrong voltage rating was erroneously installed in the NATO IIA power control unit. The discrepancy, found when the spacecraft was at ETR, was caused by improper kitting in plant. Parts control was at fault. The discrepancy could have resulted in early failure of the unit on orbit.

As a result of the findings concerning the failure of the Skynet IB ABM (see discussion above), the NATO IIB ABM was recast.

Table C.8-6. Launch Base Malfunction Summary - NATO IIA

| Malfunction | Cause | Potential Impact on Mission | Corrective Action |
|---|---|---|---|
| Batteries exceeded voltage | Battery characteristic | Minor | Deviation accepted |
| Power control unit performance not to spec. | Wrong capacitor installed | Significant | Capacitor replaced |
| Separation spring interference with thermal shield | Design error | Minor | Cups modified |
| Solar cell chipped | Improper handling | Minor | Cell replaced |
| Comm. channel power sharing out-of-limits | Design limitation | Minor | Waiver granted |

Table C.8-7. Launch Base Malfunction Summary - NATO IIB

| Malfunction | Cause | Potential Impact on Mission | Corrective Action |
|---|---|---|---|
| Battery cell overvoltage | Battery characteristic | Minor | None |
| TWT helix current out-of-limit | Cathode | Loss of redundancy | TWT replaced |
| Comm. channel power sharing out-of-limit | Design limitation | Minor | Waiver granted |
| ABM deemed unacceptable | Skynet failure | Catastrophic | Motor recast |
| ABM alignment out-of-limit | Tolerance buildup | Minor | None |

Table C.8-8. On-Orbit Malfunction Summary - NATO IIA

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 4/70 | Spin rate measurement erroneous | Improper sun sensor alignment | Minor | Sun sensor alignment changed on NATO IIB |
| 6/70 | MDA flange temp. reached qualification test levels | Thermal analysis deficiency | Minor | None |
| 6/70 | Sun angle sensor readings erroneous | Failure of data bit # 4 | Minor | None |
| 10/70 | TWT failed | Short circuit in TWT | Loss of redundancy | Switched to redundant tube |
| 12/70 | Sun angle sensor bias | Tolerance buildup | Minor | None |
| 3/71 | Telemetry transmitter high modulation index | | Minor | Range on alternate transmitter |
| 6/71 | Sun angle sensor reading erroneous | Failure of data bit # 2 | Minor | None |
| 7/71 | Earth sensor # 1 measurement incorrect | Failure of earth sensor | Minor | Antenna back-up pointing during equinox |
| 4/72 | TWT failed | Out-of-band Oscillation | Catastrophic | None |
| 9/72 | Telemetry command data erroneous | Unknown | Minor | None |

Table C.8-9. On-Orbit Malfunction Summary - NATO IIB

| Date | Malfunction | Cause | Impact on Mission | Correction Action |
|---|---|---|---|---|
| 5/71 | Sun angle sensor readings erroneous | Failure of data bit | Minor | None |
| 1/72 | MDA operation intermittent | Particle in bearing race | Minor | None |
| 6/72 | TWT degradation | Helix attenuator degradation | Loss of redundancy | Switched to redundant tube |
| 2/74 | Spacecraft rejected commands on side # 1 | Processor circuitry | Loss of redundancy | Daily command to avoid switch by timer |
| 8/74 | Telemetry frequency shift | Spurious ground signals | Minor | SCF procedure change |
| 9/74 | Comm. output instability (for short time) | Out-of-band oscillation | Minor | None |

C.8.4.2.3 On Orbit

The first failure after launch occurred during October of 1970. The Watkins-Johnson TWTA had operated for 4250 hours. An analysis of telemetry after failure showed that TWTA power output and cathode current were zero, and the helix voltage near-zero. The helix current was somewhat below the zero current indication at an equivalent telemetry voltage of about 1.18 V. Normally, the equivalent telemetry voltage for no helix current would be about 2 V due to an offset voltage generated in the power supply. A rise in battery temperature was attributed to power consumption in the TWTA current limiter in the communications dc-dc converter, which is located on the back of equipment panel 4. The main bus current change is not consistent with the normal current limiter fold-back caused by a low-resistant short. The current should have decreased to approximately 1.45 A; instead, the current increased to 1.86 A. The higher current is attributed to a partial fold-back of the limiter resulting from a higher resistance short. This premise is consistent with the equivalent telemetry voltage of 1.18 V for the helix current since the low-voltage heater switching circuits of the TWTA power supply must be operating to generate the 1.18 V offset. The high-voltage telemetry indicates a near-zero value; there is, however, on the order of 0.1 V that cannot be explained other than to postulate that the high-voltage switches were operating, but with a near-short as a load.

The second failure was characterized by rapid performance degradation beginning on 21 April 1972. The TWT (manufactured by Varian) had operated 13,500 hours on-orbit. Measurements of ERP, receiver sensitivity, and spurious signal levels recorded during the period of the anomaly showed that the ERP decreased at a more rapid rate than the telemetered RF output power, and that the receiver sensitivity decreased 6 dB in the 20-MHz channel and 20 dB in the 2-MHz channel. This change in receiver sensitivity was also confirmed by increase in apparent noise level as indicated by the IF level sensors. Spurious signals were observed in both 2- and 20-MHz channels, which would account for the change in receiver sensitivity and IF level indications. This anomaly was concluded to be out-of-band oscillation in the TWTA. The TWTA appeared to have undergone substantial cathode emission degradation.

The third TWTA (Varian) failure occurred after 11,300 hours of orbital life in the NATO IIB satellite. Analysis of data confirmed that the degradation of the TWTA was caused by a spurious oscillation. Analysis of the Varian TWT failure indicated at least two failure processes within the tube: One was likely a random failure process; the second, a wearout or depletion type process. It appears that rigid quality control and extensive development time is required to obtain TWT long-life performance.

The NATO IIB spacecraft rejected all commands (10) during four separate passes on 12 and 13 February 1974. The fault first occurred when Command Processor No. 1 was fully powered via the command enable signal transmitted at 1312 GMT on 12 February. The fault was judged to be in the command processor circuitry because (a) the spacecraft bus current remained high (1.77 A versus a nominal 1.69 A) for several hours after the processor was enabled during the four command passes and (b) the panel temperature on which the dual-command processor unit was mounted increased following rejected commands. The data did not permit the conclusive isolation of a fault to a specific component in the processor.

The NATO IIA spacecraft experienced two sensor anomalies of interest.

The first anomaly involved Earth Senor No. 1. The anomaly, which was discovered on 6 July 1971, occurred while no telemetry was being recorded; therefore, no dynamic failure data were available.

The "104" binary coded decimal (BCD) reading on 5 July was normal. The "0" BCD reading on 6 July was incorrect. All subsequent readings were erratic and incorrect. The incorrect data exhibited one of three characterisitcs during a typical 10 min interval:

a. Constant "0" BCD - earth width reset, but not counting;
b. Constant at non-zero BCD - earth width counter not reset;
c. Continuous cycling - earth width counter not stopped after counting initiated.

Coincident with the occurrence of anomalous data, the communication antenna control reference switched to Earth Sensor No. 2 from Earth Sensor No. 1.

The earth sensor was procured from Lockheed by Philco-Ford during the period of December 1968 through February 1969. The earth sensor electronics were assembled on six printed circuit boards requiring between 150 and 200 solder connections. Several earth sensors were returned to Lockheed for noise problems. Failure analysis revealed the noise problem was due to cold solder joints. Further investigation by program office (SPO) personnel revealed that several earth sensors with solder joints not meeting the Lockheed soldering criteria were allowed to pass all in-process inspections and were on delivered flight hardware. Although an in-depth analysis was not made of the anomaly, it could be hypothesized that the failure may have resulted from a poor solder joint.

The second anomaly involved Sun Angle Sensor No. 2. Analysis of the telemetry data from the sensor resulted in the conclusion that a second "bit failure" had occurred. If the data bits for the second and fourth most significant bits (64 and 16 decimal counts) were corrected, the resulting sun angles would be as expected.

The probable cause of the sun sensor failure is a flip-flop circuit or the silicon solar cell. The suspected flip-flop is in the pre-amp output register. The failure is either in wiring or a 2N2907A transistor.

In summary, the cause of earth sensor failure was not analyzed in depth by the contractor or the SPO, but it can be hypothesized that the failure could have occurred from a poor solder joint. The effect of the failure was to switch to the redundant earth sensor assembly without any mission degradation. The cause of sun sensor failure is hypothesized to be a part failure or defective wiring. The failure of the sun angle sensor did not result in a catastrophic mission failure.

The nature of these two anomalies leads to the following recommendations:

a. For critical items, source inspection should be required.

b. Customer review of build-up records, tests data, failure reports, etc., should be accomplished in a timely manner, i.e., prior to installation on flight hardware.

c. A formal failure reporting and analysis system for on-orbit failures should be included in all contracts.

References

C. 8-1. Quality Assurance Working Group Report, Air Force Space and Missile Systems Organization dated (August 1972).

C. 8-2. Skynet B Failure Review Board Final Report, Air Force Space and Missile Systems Organization (9 October 1970).

C. 8-3. IDCSP/A - Skynet Initial Launch and Orbital Status Report, Launch No. 1, TOR-0066(5135-01)-4, The Aerospace Corporation, El Segundo, Calif. (10 December 1969).

C. 8-4. IDCSP/B - Skynet Initial Launch and Orbital Status Report, Launch No. 2, TOR-0059(6135)-4. The Aerospace Corporation El Segundo, Calif. (26 October 1970.

C. 8-5. NATO Communication Satellite Initial Launch and Orbital Status Report, Launch No. 1, TOR-0066(5141)-6, The Aerospace Corporation, El Segundo, Calif. (30 April 1970).

C. 8-6. NATO Communication Satellite Initial Launch and Orbital Status Report, Launch No. 2, TOR-0059(6141-01)-8, The Aerospace Corporation, El Segundo, Calif. (17 March 1971).

C. 8-7. NATO Communication Satellite Monthly Status Reports, The Aerospace Corporation, El Segundo, Calif. (May 1970 to January 1975).

C. 8-8. IDSCP/A UK-1 Spacecraft (S/N 005), Preliminary Orbital Test Report, RN-DA2170, Philco-Ford Corporation, Palo Alto, Calif. (26 January 1970).

C. 8-9. IDCSP/A SK-1 Spacecraft (S/N 005), Final Orbital Test Report, TR-DA2181, Philco-Ford Corporation, Palo Alto, Calif. (17 April 1970).

C. 8-10. IDCSP/A Spacecraft No. 2 (S/N 006), Orbital Test Report Preliminary Final Analysis, RN-DA2198, Philco-Ford Corporation, Palo Alto, Calif. (19 October 1970).

C. 8-11. S/N 007 NATO Spacecraft No. 1 Preliminary Orbital Test Report, RN-DA2139, Philco-Ford Corporation, Palo Alto, Calif. 4 May 1970).

References (cont.)

C. 8-12. S/N 007 NATO Spacecraft No. 1 Final Orbital Test Report, RN-DA2196, Philco-Ford Corporation, Palo Alto, Calif. (20 September 1970).

C. 8-13. Preliminary Orbital Test Report For 13 Nov, Spacecraft (S/N 008), WDL-TR446, Philco-Ford Corporation, Palo Alto, Calif. (17 March 1971).

C. 8-14. Final Orbital Test Report for 13 Nov, Spacecraft (S/N 003), WDL-TR4652, Philco-Ford Corporation, Palo Alto, Calif. (2 August 1971).

## C.9 SATELLITE PROJECT A

### C.9.1 Program Summary

The Phase I part of this program included several flight spacecraft, all of which have been fabricated and launched. A Phase II modification of the satellite design has been implemented and additional spacecraft have been fabricated and acceptance tested but have not been launched. This activity is under the direction of the Air Force Space and Missile Systems Organization (SAMSO) and The Aerospace Corporation of Los Angeles, California.

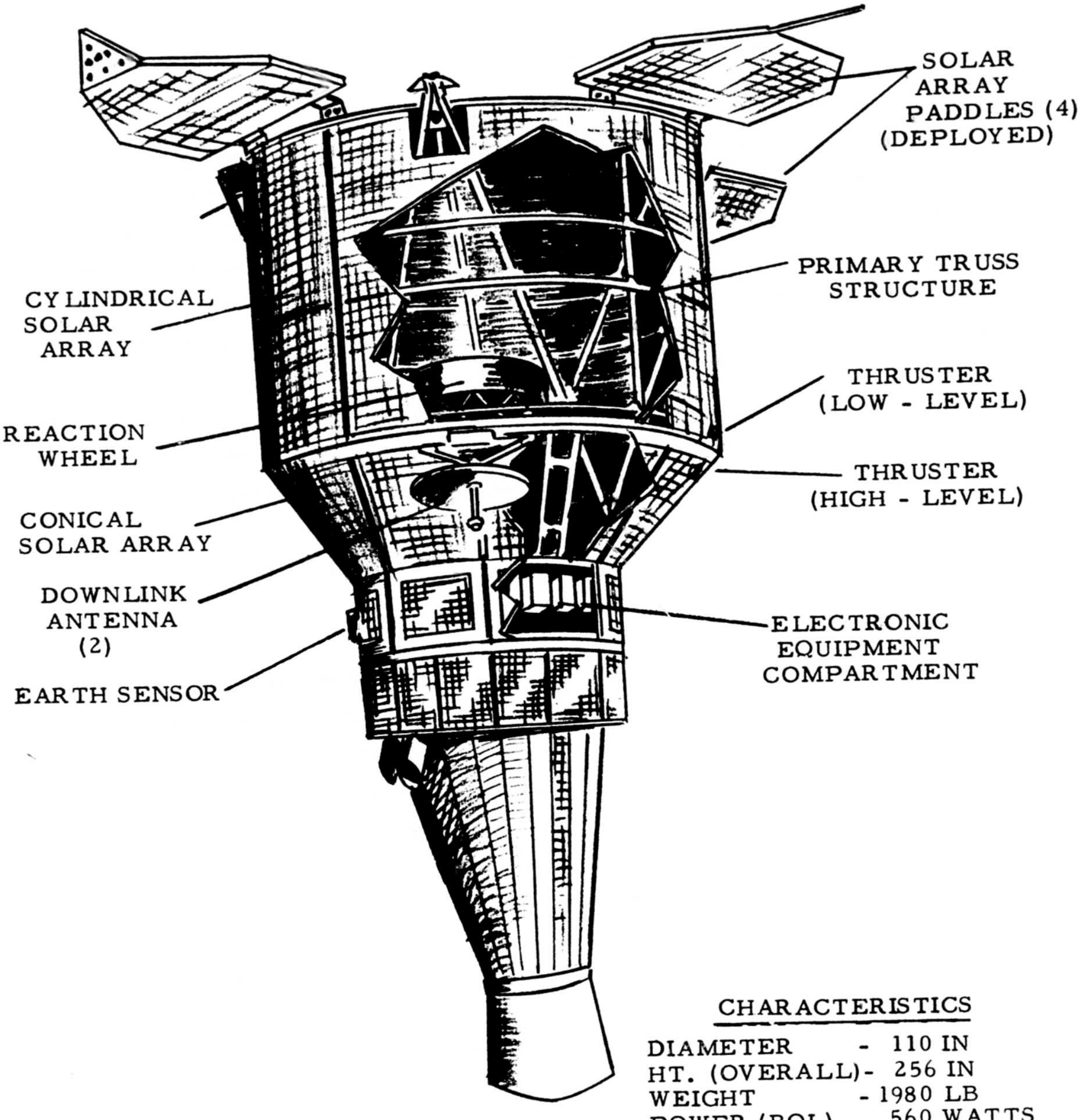
The Phase I satellites were launched from Cape Kennedy by a Titan IIIC booster and injected into a synchronous orbit. Ground communication and data processing are being provided by ground data centers. Additional information is given in Appendix D (Volume III).

### C.9.2 Satellite Description

#### C.9.2.1 General

The satellite consists of two major segments, a spacecraft segment and a primary sensor segment. The spacecraft segment provides structural support, power, and communication and command services to the sensor segment, which is attached to the top of the electronic equipment compartment. Figure C.9-1 shows a view of the Phase I spacecraft segment. The Phase II design configuration is essentially the same.

The Phase I satellites have a diameter of 110 in., a height of 256 in., and weigh 1980 lb. In addition to the primary sensor payload, two secondary payloads were flown. The spacecraft segment contains approximately 25,000 electrical piece parts excluding solar cells. The sensor segment contains approximately 110,000 electrical piece parts excluding memory cores and connectors. Reliability improvements and increased survivability were incorporated into the Phase II satellites and only one secondary payload was incorporated. The Phase II satellites have the same diameter and height as the Phase I satellites, but weigh 2240 lb.




CHARACTERISTICS

DIAMETER - 110 IN
HT. (OVERALL) - 256 IN
WEIGHT - 1980 LB
POWER (BOL) - 560 WATTS
DESIGN LIFE - 3 YEARS

BOOSTER - T-IIIC
ORBIT - SYNC. EQUATORIAL

Figure C.9-1. Project A Phase I Satellite

### C.9.2.2 Structure

The basic structure consists of a tubular truss and the main electronic equipment compartment. The equipment compartment is 18 in. high and has a diameter of 64 in. with provisions for mounting equipment to the main platform and to the external access doors. The external surface of the equipment bay is covered with a mosaic pattern of insulation and second-surface mirrors to provide an adequate thermal environment for the electronic equipment. The propulsion system is mounted on a honeycomb platform that is parallel to the equipment platform. The reaction wheel is mounted to a conical frame that surrounds the propellant tanks.

The solar array consists of a cylindrical shell, 108 in. in diameter and 79 in. long, with the array base attached to the tubular struts at the booster adapter section. A conic section 21 in. high tapers from the 108-in. diameter to the 64-in. octagonal equipment compartment. Four deployable solar paddles are attached to the base array of the spacecraft.

### C.9.2.3 Communication and Command

The communication and command subsystem transmits sensor and satellite housekeeping data to the ground stations, receives and processes the commands, and provides ranging signals for orbit determination. Two downlinks (Links 1 and 2) and one uplink (Link 3) are provided.

Link 1 includes redundant S-band 1.6-W transmitters, dc-dc converters, an RF switch for transmitter output selection, a transmitter filter, a high-gain antenna, and redundant encryption units. Link 1 performs the function of secure wideband payload data transmission at 1.024 Mbps to a satellite readout station.

Link 2 includes redundant 0.8-W transmitters, converters, an RF switch, a transmitter filter, a high-gain antenna, a dual baseband unit, two digital telemetry units for telemetry data conditioning, and redundant encryption units. The function of Link 2 is transmission to the satellite readout station of housekeeping data, some secondary payload mission data, and the ranging information. The data rate is normally 128 kbps, but can be reduced to 1 kbps. The digital telemetry unit processes the data to form a serial bit stream. The active transmitter output can be commanded to either the Link 2 high gain antenna or the Link 3 omni antenna by means of the coaxial switch.

Link 3 consists of redundant receivers, signal conditioners, converters, a diplexer, an omni antenna assembly, and a dual command decoder. Decryption is performed by redundant units. Link 2 is the reception channel for commands and ranging information. A total of 240 authenticated and 16 non-authenticated discrete commands may be processed. Commands are decrypted and decoded before they are issued to their final destination for execution.

### C.9.2.4 Electrical Power and Distribution

The electrical power and distribution subsystem is comprised of the solar cell array, power control unit, four shunt element assemblies, three batteries, the electrical integration assembly, the auxiliary integration assembly, and cabling. The subsystem provides electrical energy from solar array and battery sources, distributes it through the satellite, monitors its own operation, and provides signals for telemetry.

The electrical integration assembly contains the control circuitry needed to condition certain command signals generated in the dual command decoder, and also signals generated in other subsystems. It receives and conditions 52 commands.

The auxiliary assembly contains the controls to switch redundant encryption units and associated communication equipment into the communication and command subsystem.

### C.9.2.5 Attitude Control

The primary function of the attitude control subsystem is to acquire the earth and continually point the satellite spin axis toward the earth's center. The subsystem consists of redundant earth sensor assemblies, two sun sensor assemblies, reaction wheel assembly with redundant electronics, rate gyro assembly, redundant control electronics assemblies, redundant valve driver assemblies, redundant power converters, and a redundant equipment switching assembly. The attitude control subsystem also provides control signals to the propulsion subsystem thruster valves and gas jets which produce thrust for orbit control and torques for earth-pointing control, respectively.

The earth sensor provides earth-pointing error information in two transverse axes in the spacecraft. The sun sensor assembly consists of three silicon solar cells viewing the sun through slits in a satellite meridian plane. The sun sensors are located in quadrature on the sides of the spacecraft.

The reaction wheel assembly is spun up after spacecraft separation to a speed which causes the spacecraft to rotate at a slow rate. The result is a zero-momentum satellite. The rotagyro assembly is included in the attitude control subsystem to provide transverse body rate signals during the acquisition mode.

### C.9.2.6 Propulsion

The propulsion subsystem provides an energy source for attitude control and orbital control maneuvers. The subsystem contains 12 thrusters that are pressure-fed with hydrazine and operate in a pulse mode. Four are 3.5-lb (at 500 psia tank pressure) thrusters, and eight are 0.03-lb thrusters. Two of the larger thrusters are used for $\Delta V$ maneuver purposes and two are used for high-level attitude control during $\Delta V$ maneuvers. The smaller thrusters are used for normal attitude control. All commands to operate the subsystem are received from the attitude control subsystem via an electrical interface assembly.

## C.9.3 Satellite Orbital Experience

The first two satellites launched experienced a number of anomalies early in orbit. Many were design-oriented, which indicated the qualification program was not completely effective. Redundancy and the flexibility of the commanding system minimized the impact of the anomalies. In addition to some design changes, the factory test program was modified to increase

a. Burn-in time on components and satellites

b. Component temperature cycling

c. Vibration duration

d. Acoustic duration

In addition, all components were powered and their performance monitored during vibration test and temperature cycling.

Readiness testing every 30 days of spacecraft in storage was found to be less desirable than conducting the test only upon removal from storage for launch. Personnel errors during test caused more problems than the effect of storage.

## C.9.4 Testing and Anomalies

### C.9.4.1 Factory Test Program

The development, qualification, and acceptance test program is conducted independently by the contractors for the spacecraft segment and the sensor segment. The segments are subsequently integrated into a satellite by the spacecraft contractor, where system level qualification and acceptance tests are conducted.

Formal qualification tests for the Phase I program were conducted to demonstrate a margin of 3 dB for dynamic and 20°F for thermal beyond the maximum predicted flight environments at which acceptance was conducted. Vibration and thermal vacuum testing was conducted on most components; and humidity, acceleration, and shock were performed on a few components. The component vibration qualification was conducted for 3 min per axis at 19.5 g rms with power on to component, but key parameters not continuously monitored. The component thermal vacuum test consisted of a 12-hour soak at hot and 12-hour soak at cold predicted extremes plus 20°F margin.

System level acoustic qualification tests were performed without the flight shroud and with only the equipment operating that would normally operate during flight. Qualification tests for 3 min at 145 dB were run separately on each segment and also on the complete satellite. A satellite qualification shock test was also conducted, although the sensor segment used was the development model. Two pyrotechnic firings of the satellite separation ordnance were performed. Thermal vacuum qualification tests at the system level were not as thorough as the dynamic tests. Formal qualification testing was only conducted on the spacecraft segment since the sensor segment used in the satellite assembly was the development model sensor. Several changes were made for engineering evaluation of the spacecraft segment during the course of the test which reduced the effectiveness of the demonstration of qualification. It was an 8-day test at steady state under worst case thermal conditions using infrared lamps to provide thermal flux simulation at specific locations on the vehicle. This type of test is more to validate the thermal design rather than the qualification margin of subsystems.

Acceptance testing for the Phase I program was conducted at maximum predicted flight environments. Environmental testing at the component level consisted of three-axis vibration for 1 min at 9.8 g rms and a 24-hour thermal vacuum test with a 12-hour soak at hot and cold predicted temperature extremes. The acoustic test at 142 dB overall for 1 min was conducted at the spacecraft contractor's facility. The thermal vacuum testing of the sensor segment was conducted to align and calibrate the sensor and to check sensor performance, as well as to conduct environmental testing at flight levels in order to detect material and workmanship defects. The spacecraft segment assembly, functional acceptance testing, and acoustic acceptance testing at 142 dB overall for 1 min were conducted prior to integration with the sensor. The satellite thermal vacuum test was an 8-day test at maximum predicted flight temperatures, comprised of 3 days at cold limits, followed by 4 days of cycling between cold and hot limits (12 hours hot, 12 hours cold) and finishing with 1 day at the hot limit. This test was designed to detect material and workmanship defects.

As the result of higher than anticipated failures during the initial 45 days on orbit for the first two satellites, changes in the test program were implemented. A 30-day ambient burn-in test of the satellite was introduced starting with the third satellite. It was thought that longer duration ground testing could reduce infant mortality failures. As part of the Phase II program, additional changes were made to both the component and system acceptance testing to screen out material and workmanship defects in flight satellites. For electronic components (black boxes), the thermal and vibration tests were upgraded as shown in Tables C.9-1 and C.9-2. In addition, the perceptiveness of testing during these environments for

Table C.9-1. Component Test Improvements — Temperature-Vacuum

| | Old | New |
|---|---|---|
| • Thermal Vacuum | All | Vacuum-sensitive only (1) |
| • Temperature Cycling | None | All (2) |
| • Number of Cycles | 1 (3) | 8 (4)(5) |

(1) Transmitters, receivers, high-power components, and those containing lubricants.

(2) Includes vacuum-sensitive components.

(3) Unit powered during pumpdown if "on" during launch; 12 hours at hot temperature and 12 hours at cold; functionally tested near end of the 12-hour dwell periods.

(4) Eight cycles based upon data taken from the NASA "Long Life Assurance Study," March 1972, prepared by Martin-Marietta.

(5) Key parameters monitored continuously; complete functional test performed at the hot and cold plateaus during the 1st and 8th cycles.

Table C.9-2. Component Test Improvements — Random Vibration

Acceptance Level (> than 6 g rms overall)

| | Old | New |
|---|---|---|
| • Duration | 1 min/Axis [1] | 3 min/Axis [2] |
| • Power On | "Most" Components [3] | All [4] |
| • Continuously Monitored | No | Yes |

(1) Standard - 1/3 qualification duration.

(2) Extended in order to monitor the maximum number of key parameters without jeopardizing component life - verified by test and stress analysis.

(3) Required for all components with power on during launch and others if requested by the design engineer; pass/fail criteria nebulous and usually at the discretion of the designer.

(4) Earth sensor positioner is caged to prevent possible damage.

identifying failures, particularly intermittents, was considered improved. This was achieved by electrically energizing and operating the component, even if it will not be energized during the environment, and by continuously monitoring critical parameters. A 30-day burn-in was also implemented on transmitters and receivers. The spacecraft segment acoustic test was also increased from 1 to 2 min and for the 3-day burn-in test the sensor as well as the spacecraft segment was electrically powered.

C.9.4.2 <u>Factory to Launch Operations</u>

Following final satellite acceptance by SAMSO, the first two satellites were shipped directly to the launch pad by military aircraft. The satellite was enclosed in an air-conditioned and dynamically isolated, program-peculiar transporter that was used for road transportation as well as satellite protection during aircraft transportation. Upon arrival at the launch pad, the satellite was visually inspected and then mated to the Titan IIIC launch vehicle. After mating, a 3-day satellite integrated systems test

(IST) was conducted. The IST was in the format of a launch to on-orbit operational sequence from T-0 to T-815 min from launch. All command and data readouts during this IST were via RF from the Satellite Assembly Building (SAB) where the automatic checkout AGE was located. This program-peculiar AGE was used at both the satellite integrating contractor and the launch site. The AGE provided the capability in the automatic and semiautomatic modes for controlling the spacecraft and electrical ground equipment, continuously monitoring telemetry data, processing hardline and telemetry data, providing real-time display and data storage and generating and processing commands. The satellite was as close to flight configuration as possible with only hardlines connected for power, attitude control, and secondary payload stimuli and monitoring. Total processing time at the launch pad was approximately 30 days.

Starting with the third satellite, a period of storage in a nitrogen atmosphere followed the final satellite acceptance. Initially a satellite readiness test was conducted every 30 days, but it was found that failures were occurring due to personnel error and not to the storage environment. Therefore, a readiness test is presently conducted only when the satellite is removed from storage for shipment to the launch site or following rework.

### C.9.4.3 Flight Anomaly Summary

All on-orbit anomalies are listed in a computerized printout report (Ref. C.9-1), which is maintained current and published monthly. The data provided in this listing of anomalies are shown in Table C.9-3, and the effect of the anomaly on the mission is defined in seven categories as shown in Table C.9-4. The on-orbit anomalies experienced on four satellites are shown in Table C.9-5 by severity of the anomaly and by segment of the satellite where it occurred. Failures are defined as anomalies that seriously degrade satellite performance or life, including both primary and secondary payloads, even though redundancy has permitted recovery of operations. Categories 0, 1, 2, 3, 4 of Table C.9-4 are considered failures. Those anomalies that do not seriously affect performance or life, including

Table C.9-3. Column Headings of Anomaly Printout

| | |
|---|---|
| AN. NO. | Anomaly Numerical Identifier<br>1000 - 1999 Anomaly Numbers for Spacecraft<br>2000 - 2999 Anomaly Numbers for Sensor<br>3000 - 3999 Anomaly Numbers for Mission IIIB<br>4000 - 4999 Anomaly Numbers for RADEC |
| DEP. NO. | Vehicle number in order deployed |
| CLASS | Classification of effect on mission |
| TIME | GMT of occurrence of anomaly |
| DATE | Calendar date by month/day/year |
| DAYR | Number of the day in the year |
| SAA | Solar aspect angle |
| LONG. | Satellite subpoint longitude location |
| TITLE, OBSERVATIONS AND ACTION | Anomaly description, action taken, and reference to detailed reports |
| ANOM STAT | Open (0), closed (1), or inactive (2) anomaly status |
| FREQ/SITE | Frequency of occurrence/monitoring site |
| UNIT | Suspected anomalous unit numerical identifier |
| SIG. DATA VALUES<br>- MONITOR<br>- READING | Significant data values - telemetry monitor point and reading at time of anomaly |

Table C.9-4. Classification of Effect On Mission

| Classification | Effect on Mission |
|---|---|
| 0 | Complete loss of primary mission capability. |
| 1 | Any satellite anomaly or problem that seriously degrades the primary mission. |
| 2 | Any satellite anomaly or problem that seriously degrades the secondary mission. |
| 3 | An anomaly or problem that has a potential for degrading satellite life. |
| 4 | Any satellite anomaly that could seriously degrade satellite performance or life, for which adequate alternatives or measures were implemented. |
| 5 | Any anomaly that has a minor impact on the performance of the satellite. |
| 6 | Any anomaly that has no impact on the life or performance of the satellite. |
| 7 | Initially appeared to be an on-orbit anomaly but later analysis showed that the system performed as designed; i.e., a wrong interpretation of data, problem in ground data system, etc. |

Table C.9-5. Satellite Project A — On-Orbit Anomalies

| Orbital Days | Failures | | | | Problems | | | | Total Anomalies |
|---|---|---|---|---|---|---|---|---|---|
| | Space-craft | Pay-loads | Total | First 2 Days | Space-craft | Pay-loads | Total | First 2 Days | |
| 1229[a] | 7 | 6 | 13 | 3 | 5 | 21 | 26 | 8 | 39 |
| 1261 | 9 | 15 | 24 | 3 | 8 | 16 | 24 | 9 | 48 |
| 881 | 2 | 5 | 7 | 2 | 0 | 13 | 13 | 2 | 20 |
| 357 | 2 | 5 | 7 | 1 | 4 | 21 | 25 | 6 | 32 |

[a] Termination of satellite activity

intermittents, are defined as problems and include Categories 5 and 6. A description of each of the failures is provided for each satellite in Tables C.9-6 through C.9-9.

### C.9.4.4 Evaluation of Test Program Effectiveness

The number of failures, as well as total anomalies, occurring on the first two satellites as shown in Table C.9-5, was much higher than anticipated. The effectiveness of the ambient 30-day burn-in test of the satellite, implemented starting on the next vehicle, is shown in Table C.9-5 by the reduction of anomalies experienced on the subsequent flights. The flight failures were categorized by their effect on the mission and are plotted in Figure C.9-2. By the use of redundancy, only 37 percent of the failures resulted in a significant loss to the mission and only 6 percent were a major loss. In addition, the flexibility of the ground commanding system resulted in work-around operational modes that eliminated or reduced the impact of the failure for 38 percent of the failures. This shows the importance of adequate redundancy and flexibility in ground command. All four of these flights are considered very successful despite the high number of anomalies.

To provide more insight into the effectiveness of the ground test program, Table C.9-10 was prepared. System test effectiveness is defined as the ratio of total ground test failures to the sum of total ground plus flight failures. The increase in effectiveness for satellites 0002 and 0004 can probably be attributed in large part to the additional 30-day burn-in test; however, the large reduction in design-orientated failures would indicate the maturing of the design also had an effect. The large relative number of design-orientated failures on the earlier flights indicates that the qualification program may have not been effective. The lack of a formal thermal vacuum test of the sensor segment and the character of the thermal vacuum qualification test conducted on the spacecraft segment supports this observation. As an example, the high compartment temperatures experienced on satellite 0001 can be attributed to poor simulation of the solar heating during thermal vacuum testing.

Table C.9-6. Satellite Failures - 1229 Days On-Orbit

| Days in Orbit | Failure | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 1 | ACS mode 6 divergence | Design error in control jet firing | Loss of mission life | Pump up gas pressure by ground command; design changes next vehicle |
| 1 | Primary sensor heater inadvertent switching | Unknown | None | Ground command heater circuit |
| 2 | High equipment compartment temp. | Insufficient radiator area | Minor | Duty cycling to reduce internal power dissipation; design change next flight |
| 62 | RADEC YSS sensor stuck bit | Unknown | Potential loss of life for secondary mission | Command redundant unit |
| 137 | Link 1 temporary loss of modulation | Unknown | None | Command redundant unit |
| 304 | Primary sensor temp. rise | Degradation of solar absorptivity of radiator | Potential degradation of primary mission | None; design change made |
| 314 | Loss of primary sensor data | Power supply failure | Potential loss of primary mission | Command redundant unit |

Table C.9-6. Satellite Failures - 1229 Days On-Orbit (Cont.)

| Days in Orbit | Failure | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 418 | Primary sensor calibrate subsystem failed to turn off | Intermittent caused by insufficient motor torque | None | Command turn-off |
| 529 | ACS low level thruster valve driver failure | Transistor or line short to ground in valve driver | Reduced mission life | Use mode 5 control with higher propellant consumption |
| 636 | Excessive star sensor noise during solar flare | Solar flare | Minor reduction of mission life | Use back-up software to use sun sensor |
| 718 | Link 2A carrier intermittent | Transmitter aging | None | Work-around to turn on carrier |
| 938 | Link 2A decrease in signal strength | Amplifier tuning degradation | Potential reduction of mission life | Switch to redundant unit |
| 1229 | Hydrazine propellent depletion | Higher propellent use rate due to ACS valve failure | Termination of mission | None |

Table C.9-7. Satellite Failures - 1261 Days On-Orbit

| Days in Orbit | Failure | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 1 | Link 2 RF Switch failure | Unknown | Potential degradation of data acquisition | Operate at reduced bit rate if 2B transmitter fails; design change Phase II |
| 1 | Primary sensor defocus | Poor workmanship of Invar support rods | None | Software work-around; design change Phase II |
| 2 | Star sensor 1 sun shutter stuck open | Close clearance and contamination | Potential degradation of mission | Command redundant unit; design change next flight |
| 13 | Link 1 temporary drop in signal strength | Corona due to outgassing | None | None |
| 16 | Receiver A sensitivity degradation | Unknown | Minor | Increased ground transmitter power |
| 26 | Loss of enable signal to conduct electronics | Short to ground (intermittent) | None | None |
| 39 | Receiver A false lock-up | Intermittent failure in control circuitry | Potential reduction of life | Maintain continuous uplink lock on receiver B; design change next flight |

Table C.9-7. Satellite Failures - 1261 Days On-Orbit (Cont.)

| Days in Orbit | Failure | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 42 | RADEC bit error | Marginal shift register | Potential reduction of life | Command redundant unit; design change flight 4 |
| 43 | Spin controller malfunction | Open lead on printed circuit board or IC failure | Potential reduction of life | Command redundant unit |
| 103 | ACS mode 5 not useable | Failure of gates or wiring in CEA-B | Minor | Operate in mode 6 |
| 125 | Primary sensor temp. rise | Degradation of solar absorptivity of radiator | Potential degradation of primary mission | None; design change made |
| 126 | Primary sensor heater inadvertent switching | Unknown | None | Ground command heater circuit |
| 140 | Loss of battery C (intermittent) | Intermittent open in cell to cell wiring | Potential reduction of life | None |
| 142 | Primary sensor threshold command anomaly | Unknown | Unable to change threshold settings | Stop threshold commanding |

Table C.9-7. Satellite Failures - 1261 Days On-Orbit (Cont.)

| Days in Orbit | Failure | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 159 | RADEC ABL output change | Unknown | Potential degradation of secondary mission | Switch to redundant power supply |
| 159 | RADEC ABL lamp failure | High temp. causes evaporation of filament | Degraded secondary mission | Switch to redundant unit; design change next flight |
| 275 | Primary sensor heater failure | Unknown | Potential reduction of life | Switch to redundant unit |
| 341 | ACS inadvertent down-moding | Open circuit caused by thermal stress | None | Command to mode 6 if reoccurs; design change made |
| 349 | RADEC ABL scanner speed decrease | High bearing friction from lubricant polymerization | Minor | Use duty cycle to minimize temp. extremes; design change made |
| 431 | Mission IIIB Intermittent | Unknown | None | None |
| 457 | Excessive star sensor noise during solar flares | Solar flares | Minor - loss of some mission data | Use back-up software program to sun sensor |

Table C.9-7. Satellite Failures - 1261 Days On-Orbit (Cont.)

| Days in Orbit | Failure | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 608 | RADEC ABL power supply failure | High power operation to offset high motor bearing friction | Potential reduction of life | Use redundant unit; design change made |
| 727 | Star sensor 2 sun shutter anomaly | Close clearance and contamination | Minor | None; design changes next flight |
| 1193 | RADEC YBX calibration data failure | Cabling or connector failure | Partial loss of secondary mission | None |

Table C.9-8. Satellite Failures - 881 Days On-Orbit

| Days in Orbit | Failure | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 1 | Link 3A command receiver failure | Short circuit in converter | Potential loss of life | Command redundant unit; design change Phase II |
| 1 | Temp. difference of primary sensor radiators | Low thermal conduction at radiator connection | Potential degradation of primary mission | None; design change next flight |
| 31 | Primary sensor temp. rise | Degradation of solar absorptivity of radiator | Potential degradation of primary mission | None; design change Phase II |
| 83 | Loss of battery B | Intermittent open circuit between cells | Potential reduction of life | Operate with 2 batteries |
| 145 | RADEC ABL, scanner motor stopped | Insufficient torque to overcome high bearing friction | Serious degradation of secondary mission | Motor bearing redesign for next flight |
| 155 | Mission IIIB sensor 3 degradation | Solar flares | Partial degradation of secondary mission | None |
| 156 | Excessive star sensor noise during solar flare | Solar flares | Minor - loss of some mission data | Use back-up software program to use sun sensor |

Table C.9-9. Satellite Failures - 357 Days On-Orbit

| Days in Orbit | Failure | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 2 | RADEC intermittent fluctuations | Electromagnetic interference from intermittent connection | Degraded secondary mission | Increased sensor temp. |
| 23 | Primary sensor heater inadvertent switching | Unknown | None | Ground command heater circuit |
| 31 | Link 2B transmitter failure | Johanson capacitor, solder joint or coaxial cable | Potential reduction of life | Switch to redundant unit |
| 38 | Primary sensor temp. rise | Degradation of solar absorptivity of radiator | Potential degradation of primary mission | None; design change Phase II |
| 93 | RADEC false readings | Reflection from primary sensor sun shade | Degraded secondary mission | None |
| 261 | RADEC lamp failure | Filament failure | Potential reduction of life | Switch to redundant unit |
| 556 | Receiver A2 loss of sensitivity | Unknown | Potential reduction of life | Ground transmitter power significantly increased and commanding minimized |

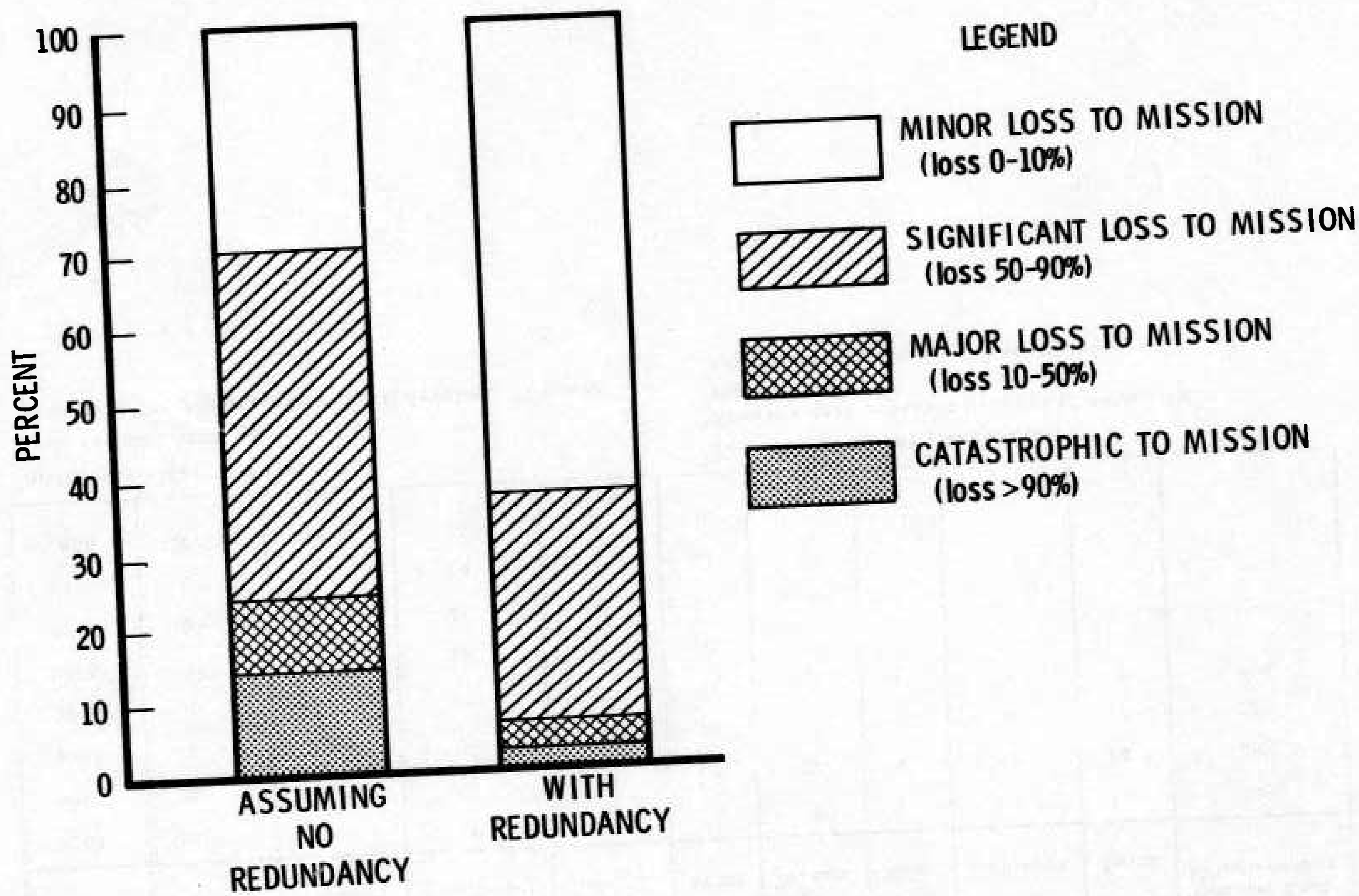



Figure C.9-2. Mission Criticality of Failures for Four Satellites

Table C.9-10. Satellite Project A – Ground Test and Flight Failures

| Satellite | System Accept Test Failures | | | | | Flight Failure Cause | | | | System Test Effectiveness |
|---|---|---|---|---|---|---|---|---|---|---|
| | Acoustic | Thermal Vacuum (8 day) | Functional | Burn-In (30 day) | Total | Design | Mfg. | Unknown | Total | |
| 0001 | 0 | 1 | 5 | - | 6 | 8 | 1 | 4 | 13 | 32% |
| 0003 | 2 | 3[a] | 3 | - | 8 | 12 | 4 | 8 | 24 | 25% |
| 0002 | 1 | 6 | 7 | 4[f] | 18 | 3 | 3 | 1 | 7 | 72% |
| 0004 | 0 | 3 | 12 | 3[f] | 18 | 3 | 1 | 1 | 7 | 72% |
| 0005 | 1[b] | 5[c,d] | 16 | 11[f] | 33 | | | | | |
| 0006 | 0[e] | 1[d] | 21 | 11[f] | 33 | | | | | |
| 0007 | 0 | 7 | 14 | 2 | 23 | | | | | |
| 0008 | 1 | 4 | 16 | 9 | 30 | | | | | |

[a] Additional 5-day T/V test
[b] Qual acoustic test
[c] Also 10 failures during spacecraft develop. T/V test
[d] 12-day T/V acoustic accept test
[e] Acoustic test repeated because of extensive replacements
[f] Additional burn-in time (6 to 21 days)

All but a very few of the functional failures during system test occurred during spacecraft integration and sensor integration with the spacecraft. The increase in such problems during later satellites is probably due to the increased attention to testing as the result of the failures in the earlier flights. The 8-day system thermal vacuum is similar to that specified in MIL-STD-1540, and the number of failures identified indicates that it is effective in identifying problems. The sharp increase in failures on satellites 0005 and 0006 is believed due to the design changes implemented in Phase II of the program, which started with satellite 0005. Only satellites 0007 and 0008 incorporated all components that were acceptance-tested to the improved vibration and thermal vacuum test program. The reduced number of failures would indicate this revised acceptance testing to be more effective. Table C.9-11 shows failures found with the old and new component test program for spacecraft segment components. Table C.9-12 shows that a higher percentage of failures are identified at the component level with the revised test procedure and failures found at the satellite level have correspondingly been reduced.

Table C.9-11. Failures - Component, Satellite (Ground), and Flight Level of Test

| Subsystem | Test Level | | | | | |
|---|---|---|---|---|---|---|
| | Component | | Satellite (Ground) | | Flight[a] | |
| | "Old" | "New" | "Old" | "New" | "Old" | "New" |
| Attitude Control | 24 | 33 | 6 | 3 | 2 | -- |
| Communications | 51 | 62 | 21 | 9 | 8 | -- |
| Electrical | 18 | 15 | 5 | 3 | 0 | -- |
| Propulsion | 50 | 21 | 3 | 1 | 1 | -- |

[a]First 45 days on orbit

Table C.9-12. "New" Test Program Results - Workmanship and Part Failures

| Subsystem | Component Failures (%) | | Satellite Failures (%) | |
|---|---|---|---|---|
| | Workmanship | Part | Workmanship | Part |
| Attitude Control | +80[a] | +116 | -50[a] | -33 |
| Communications | +17 | +54 | -67 | -50 |
| Electrical | -8 | -29 | -67 | +100 |
| Propulsion | -28 | N.C. | -67 | N.C. |

[a]Eighty percent more workmanship problems have been found at the component level and 50 percent less at the satellite level in the attitude control subsystem subsequent to the implementation of the "new" program.

## Reference

C.9-1. System Operability Update, Review, and Characteristics Evaluation (SOURCE Program Printouts), TOR-0075(5409-06)-1 through TOR-0075(5409-06)-12, The Aerospace Corporation, El Segundo, Calif. (26 July 1974 through 26 June 1975).

## C.10 DEFENSE SATELLITE COMMUNICATIONS SYSTEM - PHASE II

### C.10.1 Program Summary

The Defense Satellite Communications System - Phase II satellites (DSCS II) were designed to replace the older Phase I (IDCSP) satellites. The technology available for DSCS II is considerably advanced beyond that used for IDCSP, and thus the satellites have few similarities. It can be noted that the first launches of these two systems are separated by 5-1/2 years, the same interval of time that separated the first and fourth generation Intelsat commercial satellites. DSCS II is also known as Program 777.

The DSCS II satellite was designed and fabricated by the TRW Systems Group at Redondo Beach, California. The Air Force Space and Missile Systems Organization (SAMSO) has management responsibility for the program, with General Systems Engineering/Technical Direction provided by The Aerospace Corporation.

The DSCS II (Program 777) was granted go-ahead in March 1969, and the first launch of two satellites (Nos. 9431 and 9432) on a Titan IIIC took place on 2 November 1971. In September 1972, No. 9432 suffered a power distribution failure with loss of the despun platform and antenna pointing control, and consequently was terminated from operations. The other satellite, No. 9431, continued to operate over the Atlantic Ocean area until September 1973 when failure of the electrical power subsystem terminated the mission. A second launch for Program 777 was made on 13 December 1973, placing two satellites (Nos. 9433 and 9434) into synchronous orbit using a T-IIIC. These two satellites have carried operational traffic on a continuing basis to date (May 1975). A third launch of two more satellites is scheduled for mid-1975. (See References C.10-1 through C.10-6.)

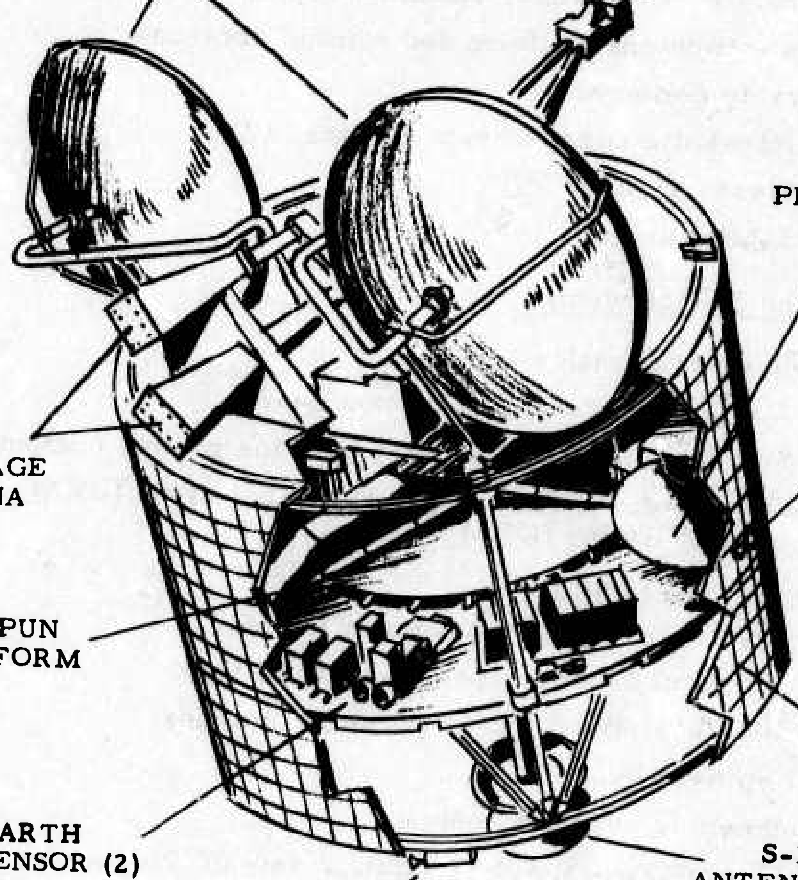
C. 10. 2 Satellite Description

Figure C. 10-1 shows the general arrangement of the satellite and describes the salient features. Each satellite weighs 1150 lb and contains 35,000 piece parts. The major items comprising the satellite subsystems are listed in Table C. 10-1.

Structurally, the vehicle consists of two spinning sections that maintain an angular velocity relative to each other. The outer cylindrical section, which is 9 ft in diameter and 5-1/2 ft high, is composed of a body-mounted solar array, an aluminum truss structure that bears the principal launch loads, and an equipment platform. The outer section spins at approximately 60 rpm, with the spin axis oriented normal to the orbital plane. The inner section of the structure consists of an earth-oriented despun platform that is isolated from the outer section by a despun bearing/motor assembly. By means of this assembly and its associated controls, the despun platform is driven so that a reference point on the platform is continuously pointed toward the center of the earth. All of the communications equipment is mounted on the despun platform. A pair of earth-coverage communication antennas transmit and receive, are fixed rigidly to the platform, and are aligned with the aforementioned reference point so that they are continuously pointed at earh center. Also mounted on the despin platform are a pair of parabolic narrow-beam antennas, each of which can be independently driven in two axes.

Command and control of the satellite is achieved by means of an onboard tracking, telemetry, and command (TT&C) subsystem, which is compatible with the Space-Ground Link Subsystem (SGLS) located at Sunnyvale, California. Most of the components of this S-band equipment are located on the spinning platform; however, certain elements, chiefly those associated with the communications and attitude control functions, are contained on the despun platform. The TT&C antenna is deployed from the aft end of the satellite and is attached to the spinning section.

Attitude control and stationkeeping of the satellite are achieved by the operation of a hydrazine monopropellant system, the 3.8 lb thrusters




LAUNCHES

#9431} #9432} - 2 NOV 71

#9433} #9434} - 13 DEC 73

BOOSTER - T-IIIC

ORBIT - SYNC. EQUATORIAL

CHARACTERISTICS

| | |
|---|---|
| DIAMETER | - 9 FT |
| HT. (OVERALL) | - 13 FT |
| WT. (LIFTOFF) | - 1150 LB |
| POWER (BOL) | - 535 WATTS |
| DESIGN LIFE | - 5 YEARS |

Figure C.10-1. Defense Satellite Communications System (DSCS II)

Table C.10-1. DSCS II Subsystem Description

Structure

- Cylindrical solar panel substructure
- Aluminum truss primary structure (spins 60 rpm)
- Despun equipment platform and comm. antennas
- Deployable components
    - Parabolic narrow-beam antennas (2)
    - Mass balance
    - S-band antenna

Attitude Control and Stationkeeping

- Attitude determination
    - Earth sensors (2), redundant heads
    - Sun sensors (2), telemetry only for ground computations
    - Electronics - despin (DEA), gimbal drive (GEA), control timing (CTA)
- Antenna pointing control
    - Despin mechanical assembly (DMA)
    - Nutation damper - pendulum/fluid type
    - Biaxial drive for each paralolic antenna
- Initial spin-up rockets (2)

  Stationkeeping reaction control
    - Thrusters - 3.8 lb (2 axial, 2 radial, 2 spin control)
    - Hydrazine tanks (4 interconnected)

Electrical Power

- Solar array - cylindrical body-mounted
- Batteries (3) - NiCd, 12.9 A-hr each
- Power control and power distribution
    - Shunt regulation
    - Automatic temperature control battery charge
    - Electrical integration ass'y (EIA) and switching logic ass'y (SLA)

Table C.10-1. DSCS II Subsystem Description (Cont.)

Thermal Control

- Passive thermal control
  - Insulation, second surface mirrors, thermal isolators
- Active thermal control
  - Thermostatically controlled heaters

Tracking, Telemetry, and Command

- S-band telemetry and command - SGLS compatible
  - Secure command and TLM links
  - Redundant receivers and decoders (cross-strapped)
  - Redundant transmitters and encoders (cross-strapped)
  - Switching logic assembly (SLA)
  - Antenna - torroidal beam omni on spinning body

Communications

- Four channels with 50- to 185-MHz bandwidths, single conversion
- Capacity
  - 1300 two-way voice circuits or ~100 Mbps digital data
- Antenna
  - Two earth coverage horns, 1 transmit and 1 receive, 18° beamwidth, 16.8 dB gain minimum
  - Two narrow-beam parabolas, 44 in. diameter, 2.5° beamwidth, 33 dB gain minimum, steerable to ±10° each axis
  - All antennas circularly polarized, mounted on despun platform

Table C.10-1. DSCS II Subsystem Description (Cont.)

Communications (Cont.)

- Transmitters - 7250 to 7375, 7400 to 7450, 7490 to 7675, 7700 to 7750 MHz
    - Four transmitter chains, one on, one standby (earth coverage), same for narrow beam - each has driver and output TWTA
    - Transmitter output - 20 W each
    - ERP - 28 dBW (earth coverage), 43 dBW (one narrow-beam antenna), 40 dBW (two narrow-beam antennas)
- Receiver - 7900 to 7950, 7975 to 8100, 8125 to 8175, 8215 to 8400 MHz
    - Tunnel diode preamplifiers and amplifiers/limiters (TDAL)
    - Noise figure - 7 dB

for which are mounted on the spinning section of the satellite in both axial and radial positions. In addition to its normal attitude control and stationkeeping functions, the onboard propulsion equipment is capable of making at least one, and possibly more, major translations of the satellite to a new longitudinal position, depending upon the velocity at which these maneuvers are performed.

The primary source of power for the satellite is a body-mounted solar array composed of NP solar cells attached to the spinning structure. This array generates 535 W of usable power at launch. Three nickel-cadmium batteries (12 A-hr) provide a secondary source of power for use during eclipse.

The satellite communication system has been designed to provide maximum operational flexibility by utilizing four separate communication channels that can be interconnected to operate with earth-coverage and narrow-beam antennas in various modes. There are four possible paths or channels that an incoming signal can take, depending on its frequency. Channel 1 is a straight-through earth-coverage channel that transmits and receives via the earth-coverage antennas. Its nominal bandwidth is 125 MHz. Channel 3 is the straight-through narrow-beam channel that transmits and receives via the narrow beam and has a nominal bandwidth of 185 MHz. There are also two crossover channels: Channel 2, which receives on narrow beam and transmits on earth coverage, and Channel 4, which receives on earth coverage and transmits on narrow beam. Both of these crossover channels have a bandwidth of 50 MHz.

The effective isotropic radiated power (EIRP) through the earth-coverage transmit beam is 28 dBW for any point on the earth from which the elevation angle to the satellite is greater than 7.5°. The EIRP of a single narrow-beam antenna is 43 dBW for any point on the earth from which the line of sight to the satellite is within 1° of the narrow-beam antenna boresight axis.

When utilizing the narrow-beam channels, it is possible to transmit in one of three modes. In the normal mode, signals can be received on one narrow-beam antenna and transmitted on the other; in another mode, signals can be received and transmitted on the same narrow-beam antenna; in the third mode, signal is received on one narrow-beam antenna and transmitted on both. Any one of these modes is selectable by commanding the position of a switch in the transponder. In the case where two narrow-beam antennas are being used for transmission, the EIRP from each one drops to 40 dBW.

### C.10.3 Key Milestones and Events

The significant program milestones are shown in Figure C.10-2, which also summarizes significant malfunctions. Additional details on satellite performance and malfunctions are given below.

### C.10.4 Satellite Orbital Experience

Tables C.10-2 and C.10-3 provide a description of all anomalies on satellites 1 and 2 (Nos. 9431 and 9432), which were launched in November 1971. Operations have been terminated on both of these satellites. The most important anomaly was the inability to despin the communications platform on satellite 2 from a standby mode, which in turn was caused by a malfunction of the despun electronics assembly. This condition was the result of improper controls system design, which was corrected on subsequent satellites. It would not have been feasible to provide a ground test that could have identified this problem. The design changes implemented to correct for the omni antenna failure on satellite 1, to eliminate single-point failures in the power distribution system, and to make the radiation generator assembly less susceptible to geomagnetic substorms would also not have shown up in normal ground testing.

The total number of anomalies on these two satellites was quite large, though only 41 percent are considered significant and 52 percent were corrected by use of redundancy or ground commanding. There were seven electronic box failures that could possibly have been eliminated by a more effective ground test program as specified by MIL-1540; i.e., temperature cycling, hot vibration, and acceptance testing at maximum predicted environment.

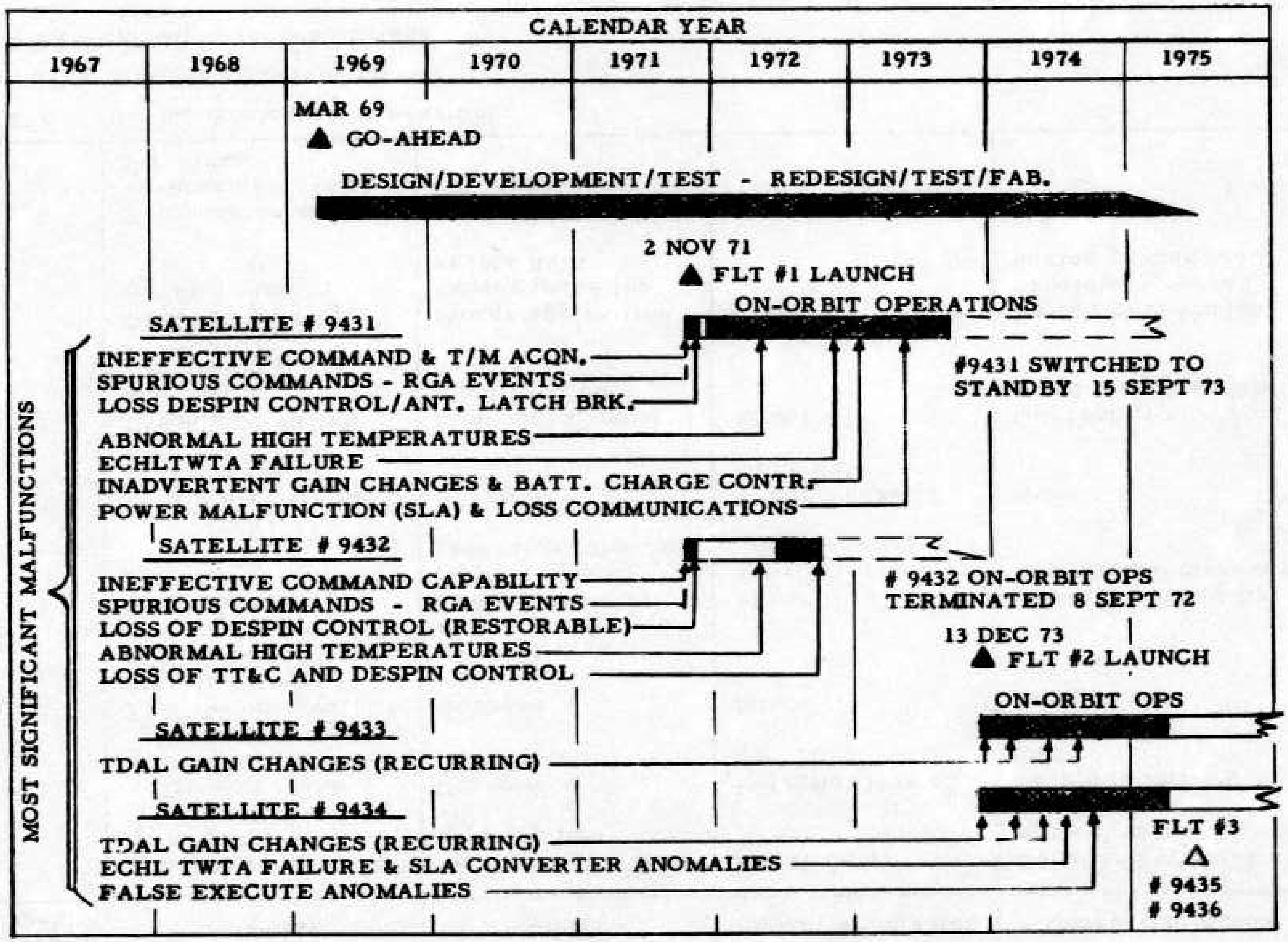



Figure C. 10-2. Key Milestones and Events - DSCS II

Table C.10-2. On-Orbit Anomalies - DSCS II, Satellite 1 (No. 9431)

| Days in Orbit | Anomaly | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 1 | Inability to process commands | Incorrect comsec key installed | Minor delay | Command address software changed |
| 1 | Telemetry transmitter failure | Unknown | Potential loss of life | Switch to redundant unit |
| 2 | EIA[a] decoder validity bit telemetry variations | Unknown | Minor | None |
| 2 | VCC[b] telemetry register scrambled | Erroneous pulse from frequency generator turn-on | Minor | Reset to normal by next ground command |
| 2 | Faulty encoder #2 channels (3) | Unknown | Minor - loss of some data | None |
| 2 | Circulator switch anomaly in commanding | Improper command identification | Minor | Correct ground command identification |
| 2 | RGA[c] #2 lock-up in reset state | Power application causes pulse that resets RGA | Potential problem | Switch to redundant unit; problem occurred during ground test |
| 5 | Temperatures reversed on radial thrusters | Sensor leads reversed | None | None |

[a]EIA - electrical integration assembly
[b]VCC - variable command count
[c]RGA - radiation generator assembly

Table C. 10-2. On-Orbit Anomalies - DSCS II, Satellite 1 (No. 9431) (Cont.)

| Days in Orbit | Anomaly | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 10 | Erroneous RGA reset commands | Geomagnetic substorms | Loss of some mission data | Design changes for next flight |
| 38 | Inadvertent switch to standby mode | CTA[d] or DEA[e] malfunction | Significant loss of capability | Command rate mode despun control using redundant CTA and DEA; rate mode convergence using earth coverage antenna |
| 38 | Telemetry and command link marginal | Omni antenna latch failures | Major - commanding difficult | Use repetitive commanding and verify at Comp Parks; design change next flight |
| 129 | Loss of telemetry (recurring) | Intermittent interruption of "on signal" during eclipse | Minor - loss of some data | Multiple commanding of transmitter "on" |
| 130 | 9°W platform bias occurred | Register scramble by unknown cause | Minor | Ground command correct bias |
| 139 | TDAL[f] gain level change occurred (first time) | Unknown | Problem in communication transmission | None |

[d]CTA - control timing assembly

[e]DEA - despin electronics assembly

[f]TDAL - tunnel diode amplifier/limiter

Table C.10-2. On-Orbit Anomalies - DSCS II, Satellite 1 (No. 9431) (Cont.)

| Days in Orbit | Anomaly | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 211 | Temp. rise on de-spun platform | Failure of forward thermal enclosure | Potential degradation | Design change next flight |
| 359 | Rapid cycling from trickle to full-rate charge | Unknown | Minor - more complex commanding | Commanded to manual trickle charge |
| 417 | Earth-coverage TWTA[g] failure | Unknown | Potential loss of some capability | Switch to redundant unit |
| 605 | No power to communication subsystems | SLA[h] commands ineffective | Termination of mission | Provide means for automated telemetry monitoring at Antigua |

[g]TWTA - traveling-wave tube amplifier

[h]SLA - switching logic assembly

Table C.10-3. On-Orbit Anomalies - DSCS II, Satellite 2 (No. 9432)

| Days in Orbit | Anomaly | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 1 | Inability to process commands | Incorrect comsec key installed | Minor delay | Command address software |
| 2 | EIA[a] decoder validity bit variation | Unknown | Minor | None |
| 2 | VCC[b] telemetry register scrambled | Erroneous pulses from frequency generator turn-on | Minor | Reset to normal by command |
| 2 | Faulty multiplexor #1 channel | Unknown | Minor-loss of some data | None |
| 2 | Circulator switch anomaly in commanding | Improper command identification | Minor | Correct ground command identification |
| 5 | CTA[c] and DEA[d] register scramble | Unknown | Minor | None |
| 19 | Erroneous RGA[e] reset commands | Geomagnetic substorms | Loss of some mission data | Design changes for next flight |
| 20 | EIRP[f] drop of 1 dB of narrow coverage channel NN-2 | Random wandering of despun platform due DMA[g] friction | Minor | Add heater to DMA bearings for next flight |

[a]EIA - electrical integration assembly
[b]VCC - variable command count
[c]CTA - control timing assembly
[d]DEA - despin electronics assembly
[e]RGA - radiation generator assembly
[f]EIRP - effective isotropic radiated power
[g]DMA - despin mechanical assembly

Table C.10-3. On-Orbit Anomalies - DSCS II, Satellite 2 (No. 9432) (Cont.)

| Days in Orbit | Anomaly | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 30 | Platform spin-up in standby mode | DEA failure | Potential loss of life | Switch to redundant unit |
| 30 | Unable to despin platform | Motor torque insufficient | Major - loss of part of mission | Several alternative methods attempted to achieve despin; successful in day 249; major despin change next flight |
| 223 | Temp. rise on despun platform | Failure of forward thermal enclosure | Potential degradation | Design change next flight |
| 304 | Battery charge control anomaly | Failed to automatically switch to trickle charge | Minor - more complex orbital operations | Over-temperature protection placed in mini-trickle charge mode |
| 307 | Power distribution failure | EIA resistor burned out due to short | Termination of mission | Design change next flight |

The flight anomalies of satellites 3 and 4 (Nos. 9433 and 9434) during 413 days in orbit are very few, with only three of significance (see Tables C. 10-4 and C. 10-5). Two of the significant anomalies are the result of spacecraft charging and arcing from magnetic substorm activity, a space environment that is not well defined and difficult to simulate in ground testing. The other significant anomaly was failure of a TWTA. The success of orbital operations compared to the first flight is attributed to both the design changes implemented and the increased testing and improved quality control as the result of the close scrutiny of all aspects of the program by the contractor and customer. However, the deficiencies in the program as compared to MIL-1540 were not corrected.

## C. 10.5 Testing

### C. 10.5.1 Factory Test Program

Component dynamic qualification tests for vibration and acoustics are conducted at the maximum predicted flight environment. The acceptance test levels for these two environments are 3 dB below the qualification test levels (the predicted flight levels). Because there is a ± 3 dB tolerance on the test levels for these two environments, it can be expected that there will probably be several frequency bands where the acceptance test levels will exceed the qualification test levels (and the predicted flight levels). Thus, there is a possibility of over-test with respect to acoustic and vibration tests. In addition, it can be expected that certain bands will not reach the desired qualification test levels, because of the ± 3 dB tolerance in testing; therefore, some bands will be under-tested in the qualification test.

All components except the solar array are exposed to the vibration qualification test and acceptance test levels. The solar array, however, is exposed to only an acoustic environment of 145 dB overall during qualification and 142 dB during acceptance.

System level qualification testing includes acoustic, pyro shock, and thermal vacuum. The system acoustic test on the assembly of two satellites in the launch configuration is conducted at 145 dB overall, which is the maximum predicted environment.

Table C.10-4. On-Orbit Anomalies - DSCS II, Satellite 3 (No. 9433)

| Days in Orbit | Anomaly | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 1 | Mass deployment telemetry switch failure | Actuator tangs deformation during ground test | Minor | Revised tooling and procedures |
| 12 | TDAL[a] gain changes and controls transients (recurring) | Spacecraft charging and arcing from magnetic substorm | Significant - loss of some mission data | Reset gain by ground command. Design changes for next flight |
| 17 | RGA[b] event | Space environment | Minor | None |

[a]TDAL - tunnel diode amplifier/limiter

[b]RGA - radiation generator assembly

Table C.10-5. On-Orbit Anomalies - DSCS II, Satellite 4 (No. 9434)

| Days in Orbit | Anomaly | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 1 | Mass deployment telemetry switch failure | Actuator tangs deformation during ground test | Minor | Revised tooling and procedures |
| 10 | TDAL[a] gain changes and controls transient (recurring) | Spacecraft charging and arcing from substorm environment | Significant - loss of some mission data | Reset gain by ground command; design changes for next flight |
| 85 | Apparent loss of gas pressure | Faulty pressure transducer | Minor | None |
| 248 | SLA[b] converter #1 turn-off | Oversensitive overvoltage circuit | Minor | Ground command to override overvoltage command |
| 287 | SLA converter #2 turn-off (recurring) | Intermittent failure | Minor | Restart converter by ground command |
| 287 | Earth-coverage TWTA[c] failure | Unknown | Potential loss of life | Switch to redundant TWTA |
| 321 | Spin-up due to extraneous command (recurring) | Sun interference | Minor | Despun by ground command |
| 413 | Sun and moon counters switched 4 hr early | Unknown | Minor | None |

[a]TDAL - tunnel diode amplifier/limiter
[b]SLA - switching logic assembly
[c]TWTA - traveling-wave tube amplifier

The thermal vacuum tests are conducted in two phases: Phase I for RF testing and Phase II for spacecraft functional testing. The Phase I test sequence is 4 days duration and includes 24-hour soaks at both the winter and summer solstice hot conditions. There is also 36-hour testing at equinox condition with two eclipse simulations. For the Phase II testing, the satellite is rotated at 50 and 60 rpm, and during the 5-day test similar conditions are simulated as in Phase I. Heat lamps are used to provide solar simulation and the outer surfaces are driven to the maximum predicted temperatures.

Component acceptance testing consists of one hot/cold temperature cycle to 10°F above and below the maximum predicted temperature extremes. Vibration tests of 1 min duration are conducted on components at 9.8 g rms overall (0.0625 $g^2$/Hz pack), which is 3 dB below the maximum predicted flight environment. Solar panels are exposed to a 142 dB overall acoustic environment for 1 min (3 dB below the qualification level).

System level environmental tests include acoustic and thermal vacuum. The two satellites assembled in the launch configuration are exposed to a 14.2 dB overall acoustic level for 1 min. The thermal vacuum test is a two-phase test of ≈ 4-3/4 days duration conducted on each satellite, which is similar to the qualification test in temperatures simulated and tests conducted with one important exception. The summer solstice condition is not simulated during the acceptance test. The summer solstice condition can be expected to produce maximum temperatures for components on the despun platform that are 10-15°F hotter than the conditions simulated.

### C.10.5.2 Factory to Launch Operations

Each satellite is installed in a protective container for shipment by C-5A aircraft to the Eastern Test Range (ETR). The satellites are shipped with ordnance installed and safed. The electrical AGE (aerospace ground equipment) shipped to ETR with the two satellites includes the communications system universal test equipment and ordnance/control and TT&C RF consoles.

Each satellite is separately tested and then mated to the other satellite in the Satellite Assembly Building (SAB) prior to transport to the pad.

The integrated systems test (IST) is performed to verify that no degradation in satellite system performance has resulted from the transportation environment. The IST was compared with the factory pre-ship IST data and the system specifications to verify that the satellite is ready to initiate the launch tasks. The IST sequence is an approximation of a normal mission profile, within the constraints of the test conditions. During a hold period in the IST, the Remote Vehicle Checkout Facility (RVCF) test is performed. The objective of the RVCF test is to verify the capability of this facility to send commands to, and receive telemetry from, the satellites. The solar array test is an illuminated test with the xenon flashlamp test set. The two satellites are mated in tandem and attached to the adapter section, which mates to the T-IIIC transtage.

After mating to the launch vehicle at the launch pad, the on-stand functional test is performed to verify the readiness of the flight at satellites for launch. Selected subsystem parameters are measured and compared to previous IST data and system specifications to ascertain that no degradation has resulted from the handling and transportation operations between the SAB and payload. The RVCF test is then performed again to demonstrate the capability of the Remote Vehicle Checkout Facility located at the SAB to send commands and receive telemetry from the DSCS II satellites when they are installed on the booster. The payload combined systems test (CST) verifies satellite ordnance circuitry for both satellites. Selected subsystem parameters are measured during the abbreviated functional test and compared to the on-stand functional test to verify subsystem integrity at a time as close to launch as feasible.

References

C. 10-1. Program 777 Orbital Operations Report, Contract F04701-69-C-0091, TRW Systems Group, Redondo Beach, Calif. (25 February 1972).

C. 10-2. Program 777 Orbital Operations Report, Contract F04701-69-C-0091, TRW Systems Group, Redondo Beach, Calif. (15 June 1972).

C. 10-3. Project 777 Orbital Operations Training Course, 9433 and Subsequent Satellites - Volume I, TRW Systems Group, Redondo Beach, Calif. (29 June 1973).

C. 10-4. Project 777 Orbital Operations Training Course, 9433 and Subsequent Satellites - Volume II, TRW Systems Group, Redondo Beach, Calif. (29 June 1973).

C. 10-5. A. T. Finney, "A Phase II Satellite for the Defense Satellite Communications System," Communication Satellites for the 70's Systems, Volume 26, MIT Press, Cambridge, Mass. (1971).

C. 10-6. System Test Objectives for Program 777, Third Flight Satellites, TOR-0075(5403-01)-3, The Aerospace Corporation, El Segundo, Calif. (19 August 1974).

## C.11 SKYNET II COMMUNICATIONS SATELLITE

### C.11.1 Program Summary

Skynet is the name of the British millitary communication satellite system. The first generation satellites (Skynet I) were launched in 1969 and 1970. One of two satellites operated successfully for several years, but the other was lost as a result of an apogee motor failure. The Skynet II satellite design is similar to Skynet I, except that the satellites are larger and heavier, and incorporate more redundancy than Skynet I to increase the design life from 3 to 5 years.

The Skynet II program was under the overall management of the British Ministry of Aviation Supply (MAS), with the U.S. Air Force Space and Missile Systems Organization (SAMSO) responsible for management of launch vehicle procurement, telemetry ground station modifications, and launch and orbital interfaces. The Aerospace Corporation contracted with both MAS and SAMSO to provide General Systems Engineering and Technical Direction for the program. Marconi Space and Defence Systems Ltd. of Portsmouth, England was selected as prime contractor for the satellite to provide the majority of subsystems and integrate them into the satellite. Philco-Ford Corporation of Palo Alto, California was a major subcontractor and supplied the attitude/orbit control subsystem and portions of the telemetry, tracking, and command subsystem and the communications subsystem.

Go-ahead for the Skynet II satellite contract was in October 1970, and the first satellite, Skynet IIA, was launched from the Eastern Test Range on a Delta 2313 launch vehicle on 18 January 1974. An electrical failure in the booster prior to second burn of the second stage caused the satellite to be injected into a non-nominal orbit. The satellite, in a low altitude decaying orbit, was first acquired by the Satellite Control Facility (SCF) on 23 January 1974. The satellite apogee boost motor was fired in an attempt to raise the low perigee, but analysis of tracking data, after the motor firing, indicated that the thrust was added in a non-optimum direction and that the spacecraft deorbited within one revolution.

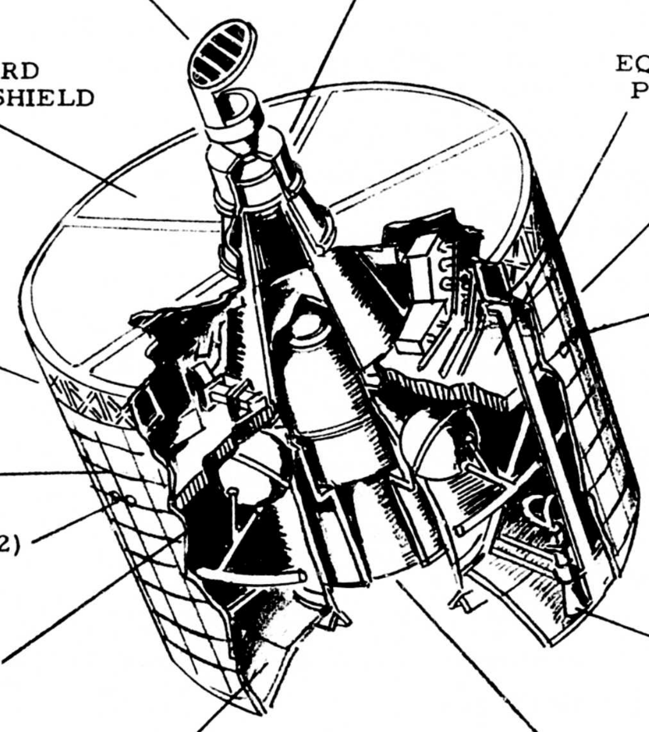
Skynet IIB was launched on 22 November 1974, and was placed into a near synchronous orbit. On-orbit operations commenced in December 1974 after the satellite had been maneuvered into its intended longitudinal position. Orbital control of the satellite was transferred from the SCF to the United Kingdom Telemetry Command Station at Oakhanger, England on 19 December 1974. Operations were continuing at the time of this report in early 1975.

## C.11.2 Satellite Description

Figure C.11-1 shows the basic configuration and key physical characteristics of Skynet II. Table C.11-1 lists the major items comprising the satellite subsystems.

## C.11.3 Key Milestones and Events

The major milestones of the Skynet II Program, including events for both Skynet IIA and IIB and most significant malfunctions that have occurred to date are given in Appendix D (Volume III). The detailed description of in-flight malfunctions and a brief listing of malfunctions that have occurred during prelaunch ground tests at the Eastern Test Range are also presented in Appendix D.




LAUNCHES
SK-IIA - 18 JAN 1974
SK-IIB - 22 NOV 1974

BOOSTER - DELTA 2313

ORBIT - SYNCH. EQUATORIAL

CHARACTERISTICS

| | | |
|---|---|---|
| DIAMETER | - | 75 IN |
| HT. (OVERALL) | - | 82.3 IN |
| WT. (LIFT OFF) | - | 960 LB |
| POWER (BOL) | - | 260 WATTS |
| DESIGN LIFE | - | 5 YEARS |

Figure C.11-1. Skynet II Communications Satellite

Table C.11-1. Skynet II Subsystem Descriptions

Structure

- Cylindrical solar panel substructure (fiberglass)
- Aluminum cone with honeycomb equipment platform (primary structure)
    - Rocket motor housing
- Despun communications antenna

Attitude and Orbit Control

- Attitude determination
    - Earth sensors (4), one pair redundant
    - Sun sensors (2)
    - Electronics - redundant
- Antenna pointing control
    - Motor drive assembly (MDA)
    - Passive nutation damper
- Reaction control
    - Thrusters - 4 lb nominal (2 axial, 2 radial)
    - Hydrazine tanks - 4 interconnected, 50 lb total
- Apogee boost motor
    - Thiokol TE-M-604-1, total impulse = 126,250 lb-sec

Electrical Power

- Cylindrical array (6 panels)
- Batteries - NiCd (2), each 12 A-hr
- Power control unit
    - Shunt regulation
    - Boost converter
- Distribution - Main, comm. control and pyro buses

Table C.11-1. Skynet II Subsystem Descriptions (Cont.)

Tracking, Telemetry, and Command

- S-band telemetry and command – SGLS compatible
  - Circular multielement antenna spins with satellite
  - Redundant command receivers plus 2 decoders (cross-strapped)
  - Redundant TLM transmitters plus 2 encoders (cross-strapped)

Thermal Control

- Passive thermal control
  - Multilayer insulation – cone, rocket motor, RCE tanks
  - Aft thermal blanket – multilayer insulation
  - Forward thermal blanket

Communications

- Two-channel X-band, 2 and 20 MHz bandwidth repeaters
- Capacity
  - Total of 24 data channels (2400 bps) or 280 voice channels
- Antenna
  - Mechanically despun horn, earth coverage, peak gain (transmit) – 18.7 dB, peak gain (receive) – 19.9 dB
- Transmitters – 7257.3 to 7259.3 and 7266.4 to 7286.4 MHz
  - Two TWTs (one on, one standby)
  - TWT rated at 20 W, operated at 18-W output
- Receivers – 7976 to 7978 and 7985.1 to 8005.1 MHz
  - Noise figure – 9 dB

## C.12 FLEET SATELLITE COMMUNICATIONS SYSTEM (FLTSATCOM)

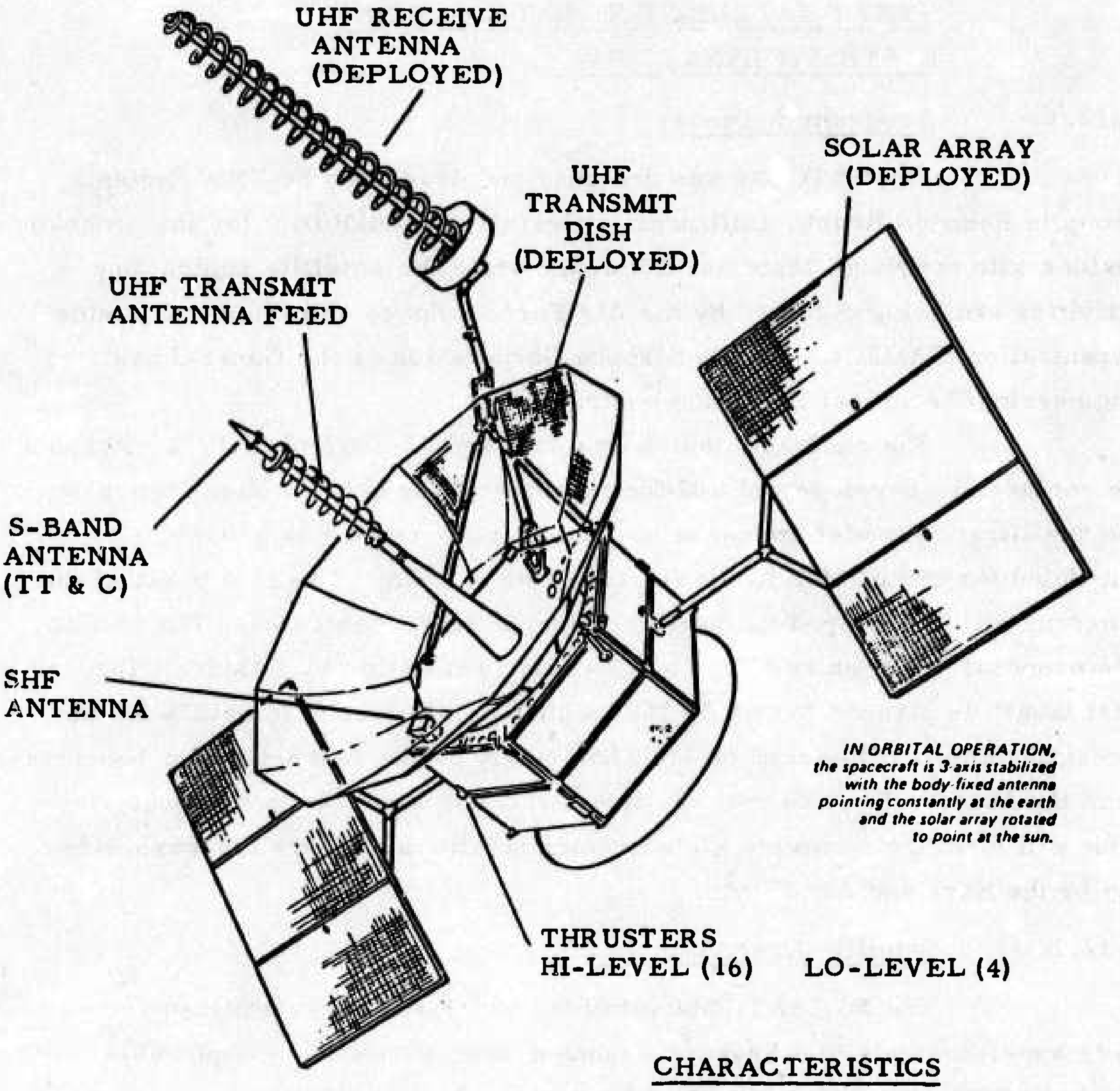
### C.12.1 Program Summary

FLTSATCOM was designed and developed by TRW Systems Group in Redondo Beach, California. Overall responsibility for the program resides with the Naval Material Command while the satellite contracting activities are being directed by the Air Force's Space and Missile Systems Organization (SAMSO). The Aerospace Corporation is the General Systems Engineering/Technical Direction contractor.

The contract, which was awarded 13 November 1972, includes the engineering development and the fabrication and qualification testing of one qualification model spacecraft. Qualification testing is presently scheduled for completion in the first quarter of 1976. The next phase of the program will be the production phase of up to seven satellites. The production proposal has been received and is under evaluation by SAMSO. The first launch is planned for early 1977 with 3-month launch intervals for the remaining units. They will be launched singly by the Atlas/Centaur boosters from the Eastern Test Range. Four satellites in a synchronous equatorial orbit will form the complete global communications network for worldwide use by the Navy and Air Force.

### C.12.2 Satellite Description

The FLTSATCOM satellite, which is body-stabilized about three axes, consists of a hexagonal-shaped body with a 16-ft deployable parabolic UHF transmit antenna and a deployable helical UHF receive antenna mounted adjacent to the parabolic dish (see Figure C.12-1). Electrical power is supplied by two symmetrical deployable solar arrays mounted on booms extending from the hexagonal body. Prior to deployment stages, the solar array panels fold around the spacecraft, forming the hexagonal




LAUNCH DATE - FIRST QTR '77
BOOSTER - ATLAS/CENTAUR
ORBIT - SYNC. EQUATORIAL

CHARACTERISTICS

| | |
|---|---|
| WIDTH (ASCENT) | - 8.7 FT |
| " (DEPLOYED) | - 43.4 FT |
| HEIGHT (ASCENT) | - 16.7 FT |
| " (DEPLOYED) | - 21.6 FT |
| WT. (LIFT OFF) | - 4100 LB |
| POWER (BOL) | - 1800 WATTS |
| DESIGN LIFE | - 5 YEARS |

Figure C.12-1. Fleet Satellite Communications System (FLTSATCOM) body in a spin-stabilized configuration for apogee kick motor (AKM) firing in the transfer orbit.

The spacecraft body is comprised of two modules. A spacecraft equipment module contains the attitude and velocity control subsystem including the solar array drive; the tracking, telemetry, and command (TT&C) subsystems; electrical power subsystem; and the AKM. The payload module houses the mission-oriented UHF and SHF communications equipment and their antennas. The Navy is provided with nine channels for relay of fleet communications and one channel for fleet broadcast; the Air Force has one wideband and 12 narrow-band UHF channels for high-priority ground-to-air or air-to-air communications. Brief satellite subsystem descriptions are given in Table C.12-1.

### C.12.3 Key Milestones and Events

A summary schedule showing key milestones and events is given in Figure C.12-2.

### C.12.4 Significant Design and Development Problems

#### C.12.4.1 Intermodulation Products

A fundamental discovery was made during the first brassboard model testing that intermodulation (IM) products larger than the receiver threshold level could be generated by passive devices such as loose joints and sharp corners. These IM products had been identified before, but were not at a troublesome level. Two factors that were not generally found on other programs contributed to this: the first was the large frequency separation between the 23 carriers needed to allow separate transmitters followed by multicoupler filters; the second was the relatively narrow spacing between transmit and receive bands.

Considerable effort was expended in identifying the IM sources and finding solutions. The real breakthrough was made when the design shifted to separate receive and transmit antennas, providing more than 45 dB of isolation and hence reduction in IM products by this amount.

Table C.12-1. FLTSATCOM Subsystem Description

Structure and Configuration

- Hexagonal honeycomb and stringer body with two main modules – payload and spacecraft
- Solar array wraps around main body during launch and ascent
- Solar array unfolds after AKM firing at apogee

Attitude and Velocity Control

- Three-axis stabilization control
- Body-fixed antennas
- Clocked solar array drive
- Pitch and yaw axes – body-fixed momentum biased reaction wheel
- Roll – offset thrusters
- Earth sensor – 2 redundant heads (active flexure)

Reaction Control

- Pressure-fed $N_2N_4$
- Spin, despin, precession, positioning, repositioning, E-W stationkeeping orbit correction
- Attitude control - sun acquisition, earth search, $\Delta V$, normal mode (momentum wheel unloading), reacquisition

Electrical Power

- Unregulated power bus (20-70 V), converters to all loads
- Solar array - 22,920 cells (2 × 4 cm) on two wings
- Array output - BOL = 1800 W, EOL (5 yr) = 1435 W
  - Separate charge array included (200 W)
- Batteries - Three NiCd, 24 cells each, 24 A-hr each
  - Cell bypass circuitry

Thermal Control

- Passive radiators, insulation, and heaters

Table C.12-1. FLTSATCOM Subsystem Description (Cont.)

Communications

- SHF - 1 transmitter
- UHF - 12 transmitters (10 Navy, 2 AF)
  - Channel 1 - fleet broadcast (SHF/UHF)
  - Channels 2-10 - Navy relay (UHF)
  - Channels 11-22 - AF narrow-band users (UHF)
  - Channel 23 - DoD wideband users (UHF)
- Communications equipment located in separate "payload module"

Tracking, Telemetry, and Command

- S-band, SGLS-compatible
- Encrypted, clear command and telemetry
- Telemetry bit rate - 250 or 1000 bps
- PRN (pseudo-random noise) ranging and doppler

Antennas

- UHF
  - Transmit - 16-ft parabolid (backfire bifilar helix feed)
  - Receive - 18-turn aluminum ribbon helix
  - Multicoupler/filter - 13-port sealed and pressurized to prevent multipacting
- SHF - Horn antenna looking through hole in UHF mesh dish
- TT&C - Log conical spiral

Reliability

- Design life - wearout, 5 years; expendables, 7 years
- Reliability - 24 at 5 years
- Mean mission duration - 38 months
- No electronic component single-point failure modes, extensive cross-strapping and redundancy
- Ground-command backup and overrides for automatic functions

Survivability

- Meets Joint Chiefs of Staff (JCS) requirements without radiation detection, circumvention, or ground command reset techniques; also capable of unattended operations for 20 days

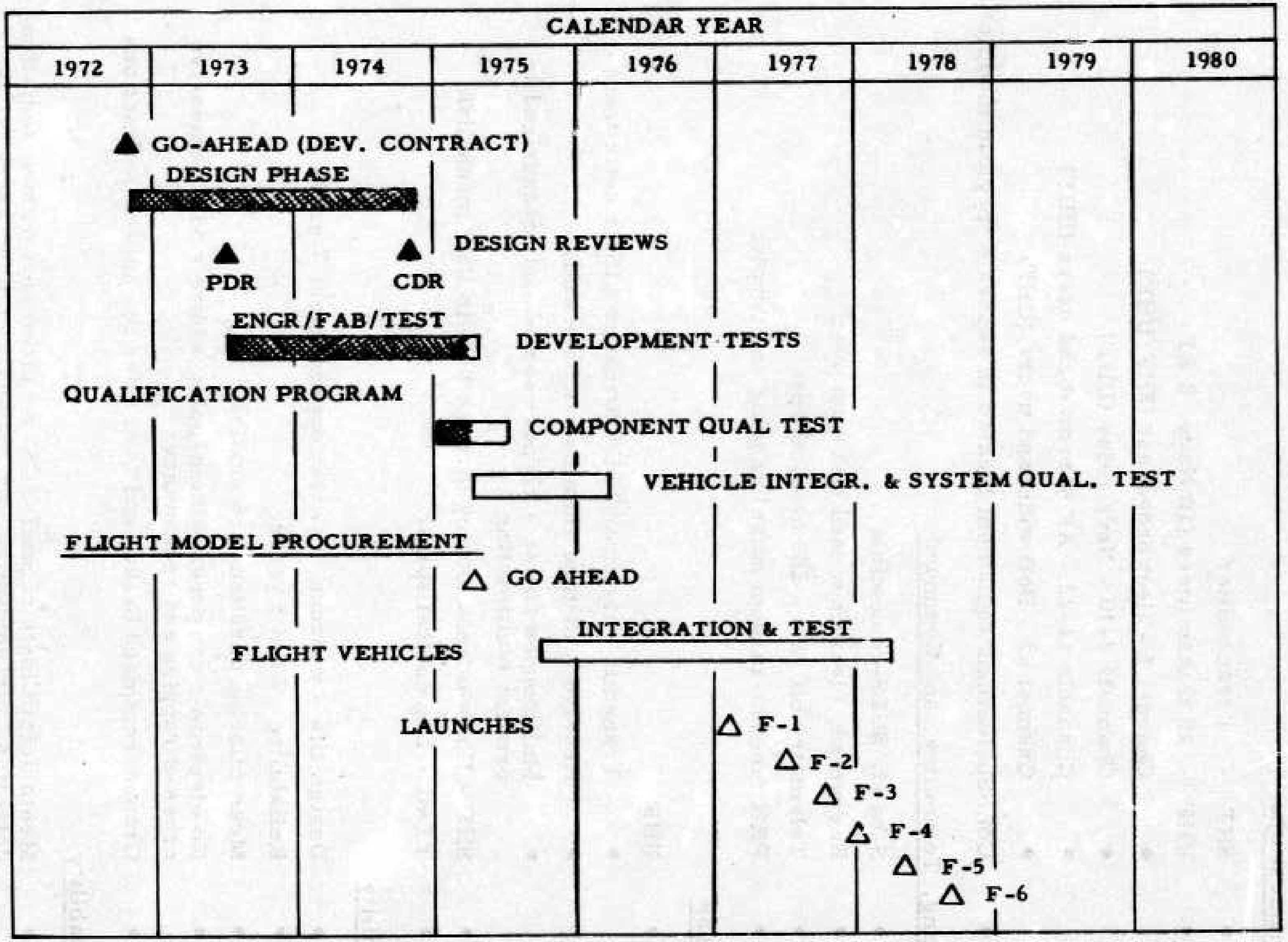



Figure C.12-2. Key Milestones and Events — FLTSATCOM

This problem was fundamental and would have occurred no matter which contractor was selected, as indicated by the presence of IMs to greater or lesser degree on nearly all other programs involving multiple transmitters using common transmitting and receiving antennas.

### C.12.4.2 Multipacting

Multipacting can occur in a vacuum whenever the proper conditions of voltage, frequency, and spacing are met. The large number of transmitters used in the FLTSATCOM program cause high peak voltage levels that make the probability of multipacting greater than when only one transmitter is used. The problem occurred during early multicarrier vacuum power tests, but was not immediately identified because of improper test disciplines. It was not until the second power handling test a year later that the multipacting problem had its full impact.

A systematic investigation was begun which resulted in redesign of the transmit feed so that the balun could be moved to a less temperature-sensitive location and pressurized or filled with teflon or both. Separate teflon-filled coaxial transformers with coaxial lines were then used to feed the two terminals of the bifilar helix.

The problem remaining is the excessive temperature expansion properties of the teflon dielectric. This is basically a combined material and mechanical problem that is made difficult by the large temperature excursions of the antenna feed elements.

### C.12.4.3 Air Force Transmitter Stability

Two problems were encountered on the FLTSATCOM program relative to stability of the Air Force transmitter. One was repeatability of intermodulation measurements and the other was break-up at cold temperatures with two carriers present.

## C.13 SPACE TEST PROGRAM (STP) 71-2 (AGENA SATELLITE)

### C.13.1 Program Summary

The STP 71-2 program was performed by the Space Systems Division of the Lockheed Missiles and Space Company (LMSC) for the Air Force Space and Missile Systems Organization (SAMSO) with The Aerospace Corporation performing the General Systems Engineering/Technical Direction (GSE/TD) role. The program comprised a single Thorad/Agena launch from the Western Test Range (WTR) with the Agena acting as both an ascent stage and as a spacecraft in support of four experiment equipments, which were government-furnished equipment (GFE) to the program.

Program go-ahead was on 1 April 1970. The Agena was shipped to the launch base on 21 September 1971 and the launch took place on 17 October 1971. Although the design mission life was 6 months, the vehicle operated and produced useful data for 26 months when the last battery failed.

The primary objectives of the program were to carry four experiments into nominal 425-mile polar circular orbit to meet flight objectives of each experiment as follows:

a. Flexible Solar Array (RTD-806) - Demonstrate solar array extension, retraction, and linkage; power transfer capability, tracking, and lock-on performance with changing beta angle; power generation capability; and solar cell performance during prolonged orbit operations.

b. Celestial IR (SAMSO-002) - Accumulate and reproduce three orbits of celestial infrared data.

c. Input-Output (ONR-001) - Accumulate data from ionosphere region; perform radar and VLF (very low frequency) propagation studies.

d. BATSON (NSA-101) - Validate BATSON operations.

Secondary objectives were as follows:

a. SAMSO-002 - Evaluate operating life of the cooling system and erection of the sensor system during mission life.

b. ONR-001 - Acquire additional data on effective electron-ion recombination rates versus altitude in E- and F-region of ionosphere.

## C.13.2 Satellite Description

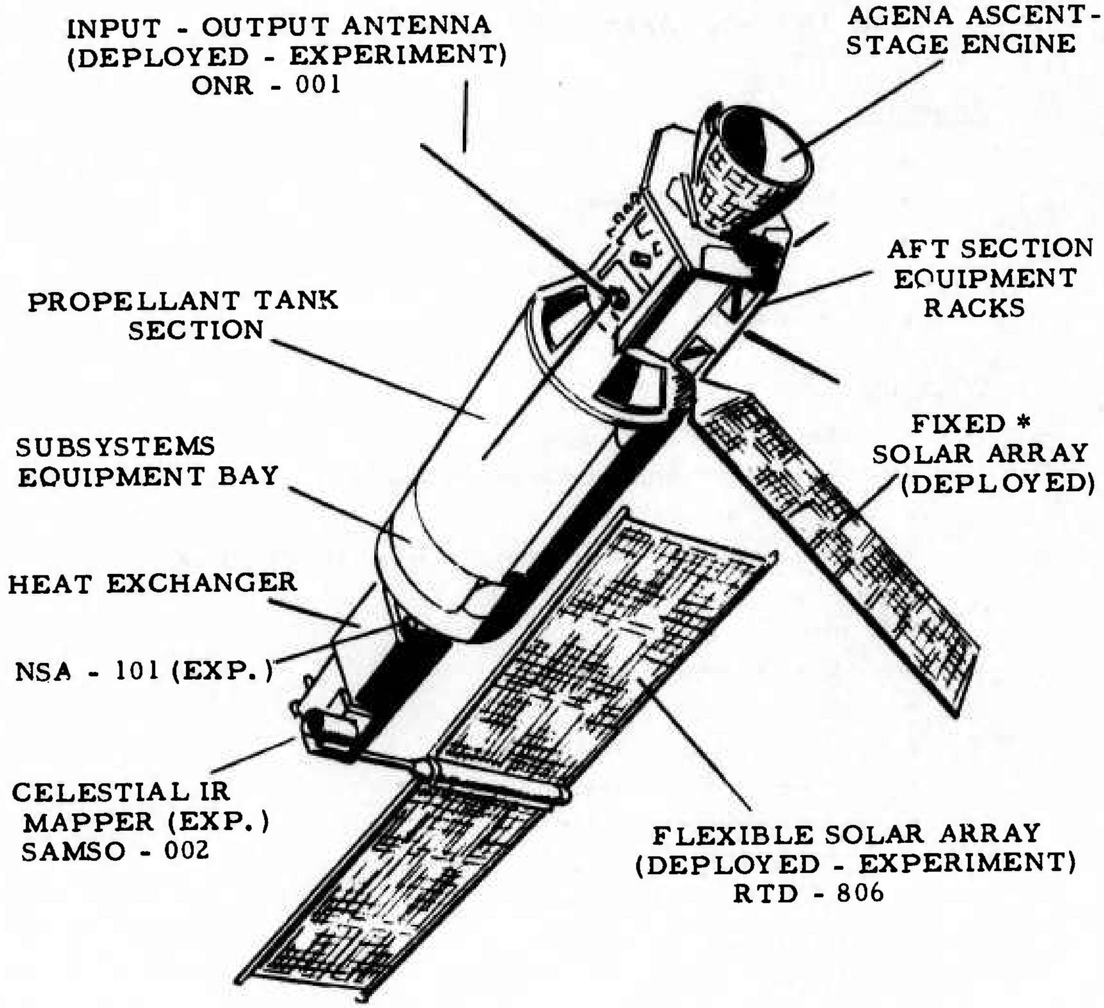
The satellite was a standard Agena with one experiment located on the aft rack and three experiments on a forward payload module. The satellite was vertically oriented in orbit as shown in Figure C.13-1. The overall spacecraft dimensions prior to deployment of the solar array and experiments were a diameter of 60 in. and length of 31.5 ft. The on-orbit empty weight of the satellite was 3443 lb. The Agena subsystem components are listed in Table C.13-1. System and subsystem data were obtained from Reference C.13-1.

## C.13.3 Key Milestones and Events

The key milestones through launch were:

| | |
|---|---|
| Program Go-Ahead | 4/1/70 |
| Preliminary Design Review | 9/18/70 |
| Critical Design Review | 2/11/71 |
| Vehicle CART | 5/21/71 |
| Ship to Vandenberg AFB | 9/21/71 |
| DD-250 | 9/25/71 |
| Launch | 10/17/71 |

Significant program events and malfunctions are summarized in Figure C.13-2.




LAUNCH DATE

17 OCT 1971

BOOSTER - THORAD/AGENA

ORBIT - 425 NM
$93^{o}$ INCLINATION

CHARACTERISTICS

| | | |
|---|---|---|
| DIAMETER (ASCENT) | - | 5 FT |
| WIDTH (DEPLOYED) | - | 40 FT |
| LENGTH (ASCENT) | - | 31.5 FT |
| WT (ON-ORBIT) | - | 3443 LB |
| * POWER (BOL) | - | 435 WATTS |
| DESIGN LIFE | - | 6 MONTHS |

Figure C.13-1. Space Test Program (STP) 71-2 (Agena Satellite)

## Table C.13-1. Agena Subsystem Components

**Structures**

- Aft rack
- Propellant tank assembly
- Forward rack
- Booster adapter
- Nose fairing

**Propulsion**

- Bell 8096 rocket engine
- Associated propellant lines, valves, and couplings
- Helium sphere
- Pressurization system lines, valves, and couplings

**Electrical Power**

- Inverter type 12A
- Inverter type 15
- Power distribution box
- Power transfer switch
- Converter type 9A
- Forward safe arm box
- Aft control and instrumentation box
- Destruct/discrete box
- Magnetic timer
- Amp-hour meter
- Current sensor
- Solar array module
- DC power and telemetry J-box
- AC power and telemetry J-box
- Fuse resistor J-box
- ONR control box
- Power control logic assembly type 1
- Power control logic assembly type 2
- Type VII secondary battery
- Type 30 primary battery (3)
- Charge controller - spacecraft bus
- Shunt regulator - SAMSO bus

Table C.13-1. Agena Subsystem Components (Cont.)

Guidance and Attitude Control

- Horizon sensor (2)
- Horizon sensor electronics
- Flight control electronics
- Inertial reference package
- Velocity meter
- Velocity meter counter
- Guidance J-box
- Flight control J-box
- Sequence timer (2)
- WECO kit
- Control moment gyros (4)
- Tri-axial, Magnetometer
- Nitrogen sphere
- Nitrogen regulator
- Thrust valve cluster (2)
- Hydraulic actuators (2)
- Hydraulic power package

Telemetry, Tracking, and Command

- PCM telemeter type 3
- KCX-28 (2)
- PCM unit type 6
- Submultiplexer type 3
- Baseband assembly unit (2)
- UHF - PM transmitter - 2 W (2)
- RF switch type 14
- UHF multicoupler type 14
- RF switches
- Antenna type 28
- Antenna type 7
- PCM MOX type 5
- Tape recorder type 32 (2)
- Transmitters (2) - 10 W
- RF couplers
- UHF receiver - demodulator type 2 (2)
- Decoder type 23 (2)
- Subdecoder (2)
- Various antennas

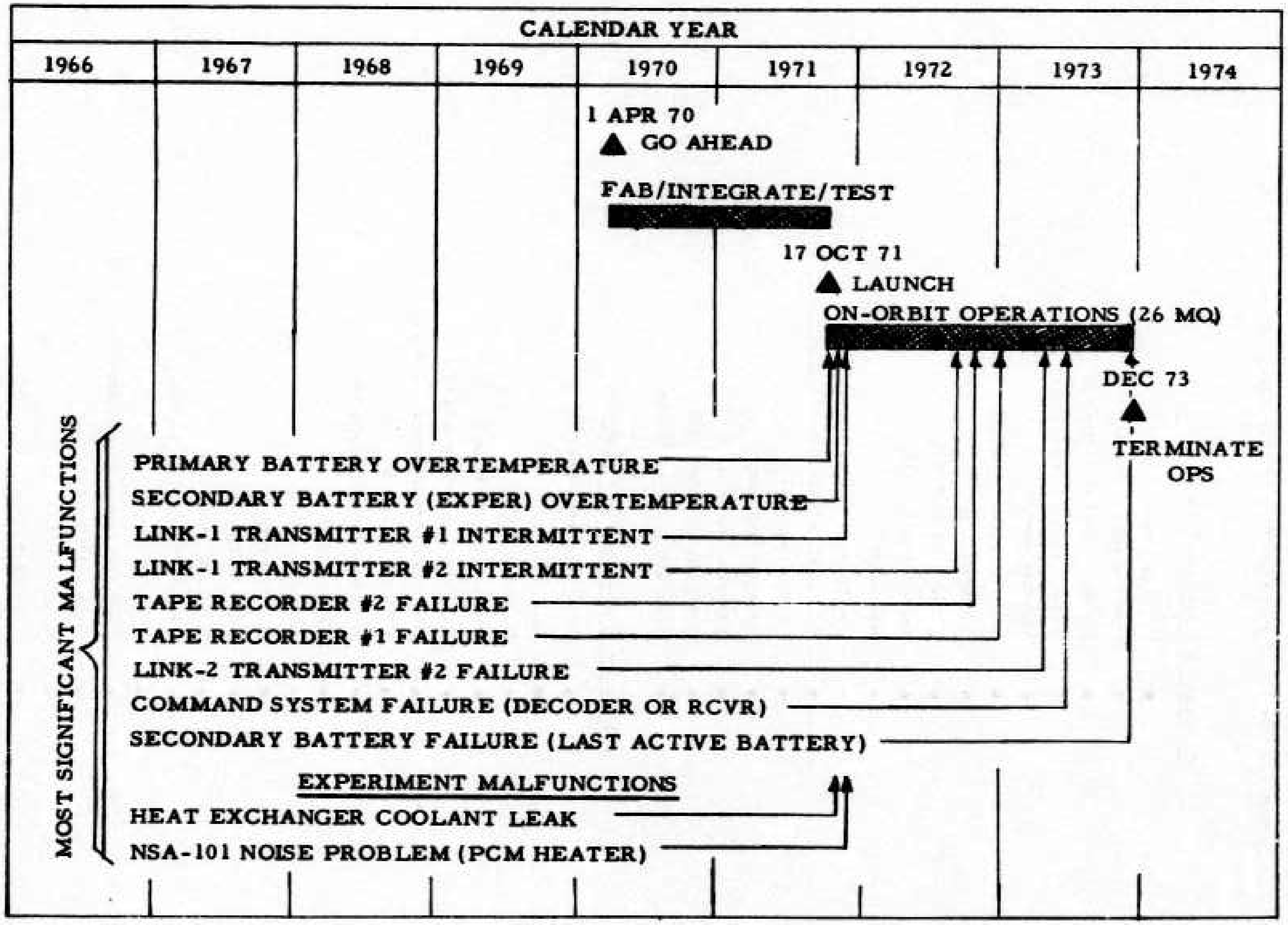



Figure C.13-2. Key Milestones and Events - STP 71-2

### C.13.4 Flight Malfunctions and Anomalies

The malfunctions and anomalies that occurred during the STP 71-2 flight are summarized below along with significant non-anomalous operations. This information was obtained from Reference C.13-2.

#### C.13.4.1 Primary Battery 1 Over-Temp.

On the first day in orbit, primary battery 1 experienced an over-temperature condition. The probable cause was either an internal short or incorrect thermal and load sharing designs.

#### C.13.4.2 Secondary Battery 2 (SAMSO Bus)

Secondary battery 2 reached 162°F on day 21. There was some damage (reduced capacity) to the battery, but it was not critical. The probable cause was a thermal/overcharge condition. The SAMSO bus (secondary batteries 1 and 2) provided back-up power until day 333 when loss of the second carrier 1 transmitter necessitated shutdown of the SAMSO bus. During operation, voltage extremes of 0.7 to 30.5 V were experienced, as well as temperature extremes of 10°F to 135°F (battery 1) and 162°F (battery 2).

#### C.13.4.3 Carrier 1 (2 W) (Spacecraft Operations - Health)

Transmitter 1 of this carrier became intermittent on day 60. Transmitter 2 became intermittent on day 333. Up to the point of mission termination both transmitters were producing intermittent data of from 1 to 750 sec (total pass). No correlation was found with temperature or voltage. One or the other transmitter was used to transmit some data on 25 percent of the passes.

The total operational times to failure for the Carrier 1 transmitters were:

a. Transmitter 1 - 100 hr [540 (12 min) contacts]

b. Transmitter 2 - 260 hr (1300 contacts)

C.13.4.4 Carrier 2 (10 W) (Experimenter and Tape Recorder)

Transmitter 1 of carrier 2 failed to produce data on day 226. Subsequently, this transmitter operated satisfactorily (day 502 to 528 and day 532 to mission termination, day 715). Failure on day 226 may have been with RF switch contact (~200 transfers to that point). Probable cause was dirt, corrosion, or a random failure to transfer. (It was during a transfer from transmitter 2 to transmitter 1 that the failure occurred. Transmitter 1 was drawing power, however.)

Transmitter 2 (550 hr) failed to produce data on day 502. It was tried again on day 528 and operated satisfactorily to day 532 when it again failed. A transfer was not involved during either of the transmitter 2 failures and it did not draw power.

C.13.4.5 SAMSO-002 Coolant Loop

An apparent leak commenced on day 13; by day 21 when the coolant system was shut down due to temperature problems on battery 2, the accumulator outlet pressure indicated that the accumulator was fully compressed, i.e., empty. An attempt to restart the coolant loop and SAMSO-002 system was initiated on day 44, but was terminated after four revolutions due to insufficient coolant flow. The most probable cause has been stated to be a micrometeoroid penetration of the heat exchanger or other tubing.

C.13.4.6 Tape Recorders

Tape recorder 2 (SR1001) failed on day 371. It appeared to be stalled at BOT (beginning of tape) and drew excessive current (~1.5 A versus 0.5 A steady state). Tape recorder 2 had previously exhibited a track-jumping condition (five occurrences) only when the "end of track 2 override" was enabled. This was believed due to susceptibility of stray voltage in check-out circuitry, which is not used in the flight mode. Tape recorder 1 (SR1002) failed during a readout when switching from track 7 to track 6 (day 445). Logic control appears to have failed. The tape recorder will not move in either direction following this failure.

The following is a summary to the tape recorder operating times:

| Serial No. | Operating Time (hr) At Odetics | LMSC | Flight | Total |
|---|---|---|---|---|
| 1001 (No. 2) | 350 | 142 | 1696 | 2188 |
| 1002 (No. 1) | 579 | 129 | 2120 | 2828 |

C.13.4.7 Command System Switch

The decoder or receiver switch associated with the U2XXX command system completely failed on day 600. It was not possible to determine the cause. No commanding was possible through this system.

C.13.4.8 NSA-101 Noise

Starting with day 50, the NSA-101 experiment began to experience a large noise input. This subsequently proved to be related to the $2F_1$ stable oscillator heater of the PCM-5 unit. Satisfactory operation was resumed with the heater power off (compensations must be made for frequency shifts). The most probable cause was a crack in the dewar.

C.13.4.9 Pneumatic Control System (non-anomalous)

This system was activated on day 40 for restabilization while the vehicle was being torqued at several times the capability of the gravity gradient/CMG (control moment gyro) system. The source of the torque, which lasted approximately 10 hours, was never determined.

The system was activated again on day 183 for approximately 90 min to reacquire the gravity gradient/CMG control. The vehicle was placed in large angle coning motion as a result of reactions from spin-up of the RTD-806 array. At that time there were 22 lb of control gas remaining (lifted off with 47.5 lb).

C.13.4.10 Horizon Sensor Operation (non-anomalous)

The horizon sensors have been operated intermittently over a 21-month period. They have been used for early orbit attitude control, recovery from unstable conditions (twice), and to check attitude behavior during RTD-806 and vehicle maneuvers (three yaw-arounds). Total operating time on orbit has been 125 hours.

C.13.4.11 Secondary Battery/Solar Array Operation (non-anomalous)

The vehicle has performed approximately 10,400 revolutions with the single secondary battery and P-69 solar array. There have therefore been approximately 15,000 battery cycles and approximately 12,000 activations of one or both of the charge controllers.

C.13.5 Evaluation of Program Experience

Some of the factors which contributed to the success of the program included:

a. Rebuilt all orbital critical equipment.

b. Tested all boxes drawn from stock; changed parts to hi-rel (in some cases); and added redundancy on back-up mode to boxes.

c. Included at least one back-up mode for every critical function.

d. Performed system level FMEA (800 hr actual).

e. Parts engineers made visits to manufacturers to preclude recurrence of problems noted during test.

f. Constant communication was maintained with subcontractors.

g. Close cooperation was maintained with Aerospace (GSE/TD contractor).

h. Test screening program was increased to 250 hr burn-in, 150 hr failure-free operation.

i. Special envelope testing was required beyond standard qualification test and fly, particularly mechanical equipment such as antennas and deployment mechanisms.

j. Life test was required on the tape recorder (extended qualification test used short time at accelerated factor).

The reliability/maintainability/safety effort on the program was approximately 4 percent of all hours (excluding manufacturing and quality assurance). The vehicle did not have an acoustic test or a thermal-vacuum test.

## References


C.13-1. SESP 71-2 Contract Item Design Review, LMSC/A971660, Lockheed Missiles and Space Company, Sunnyvale, Calif. (1 March 1971).

C.13-2. LMSC IDC from J. A. Donnelly to W. L. Finch, "SESP 71-2 Success Story," Lockheed Missiles and Space Company, Sunnyvale, Calif. (8 November 1971).

## C.14 NATO III COMMUNICATIONS SATELLITE

### C.14.1 Program Summary

The NATO III satellite is being designed and developed by the Philco-Ford Corporation of Palo Alto, California. The Air Force Space and Missile Systems Organization (SAMSO) is the managing contracting agency and The Aerospace Corporation is the General Systems Engineering/Technical Direction contractor.

The satellite contract was awarded 5 March 1973. Two flight spacecraft, one thermal/structural model and one qualification model, were included in the contract. Optional provisions were also made for refurbishing the qualification model for use as a third flight satellite. The first launch is estimated to be in early 1976. The launch vehicle is a 2914 Delta.

### C.14.2 Satellite Description

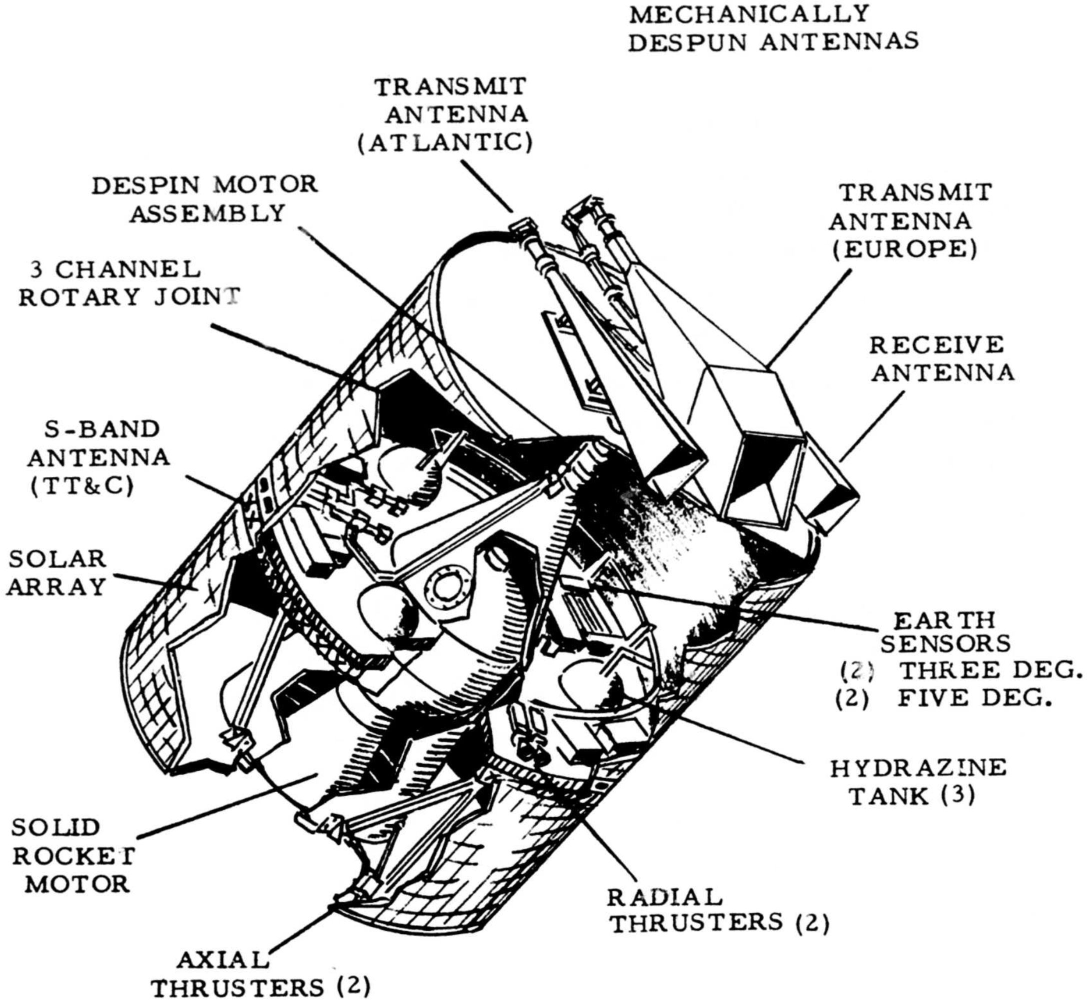
The NATO III satellite is spin-stabilized and will be placed in a 24-hour synchronous orbit. It has a cylindrical-shaped body which is covered with 20,120 solar cells (see Figure C.14-1). There are three X-band communication antennas: Atlantic transmit, European transmit, and receive. The antennas are mechanically despun to point toward earth and are fed through a three-channel rotary joint attached to the spin-stabilized section where the communication equipment is mounted. Salient satellite features are given in Table C.14-1. Brief subsystem descriptions are listed in Table C.14-2.

### C.14.3 Key Milestones and Events

An overall program schedule is shown in Figure C.14-2.

### C.14.4 Significant Problems Occurring During Program

The most significant problem that occurred during the NATO III contract was related to the difficulty the prime contractor had in getting delivery of high-reliability piece parts on schedule. First, it must be realized that the specified piece part requirements for the NATO III program emphasized extreme reliability and quality. Philco-Ford accepted these requirements. (They did not request nor would SAMSO entertain waivers to allow lower quality parts.)




LAUNCH DATE - LATE 1976
BOOSTER - DELTA 2914
ORBIT - SYNC. EQUATORIAL

CHARACTERISTICS

| | | |
|---|---|---|
| DIAMETER | - | 86 IN |
| HT. (OVERALL) | - | 121.7 IN |
| WT. (LIFTOFF) | - | 1532 LB |
| POWER (BOL) | - | 533 WATTS |
| DESIGN LIFE | - | 7 YEARS |

Figure C.14-1. NATO III Communications Satellite

Table C.14-1. NATO III Spacecraft General Features

- Spin stabilized
- Despun communications antenna
- No deployable mechanisms
- Fully operational through eclipses
- Seven-year life (expendables)
- Spacecraft commands required infrequently (once per month)
- Payload weight - 1528 lb
- Power - 540 W (equinox beginning of life)
- Size
  - Cylinder
    - Diameter - 86 in.
    - Height - 88.7 in.
  - Spacecraft overall height - 121.7 in.

## Table C.14-2. NATO III Subsystem Description

<u>Structure</u>

- Primary - central cylinder mounted on Delta 3731A adapter
- Spinning platform supported by central cylinder and struts
- Despun antenna mounted on despun mechanical assembly (DMA) at forward end of central cylinder

<u>Attitude Control</u>

- Hydrazine thrusters (redundant 5 lb axial and radial)
    - 50.0 lb propellant budget
- Sensors
    - Earth (4)
    - Sun (2)
- Nutation damper
    - Ball in tube
- Apogee kick motor (AKM)
    - Aerojet SVM-6 (6% on load capability)

<u>Power</u>

- Two-segment cylindrical solar array
- Body-mounted solar cells
- Batteries - NiCd

<u>Thermal</u>

- Primarily passive
- Heaters (hydrazine, AKM, battery)

<u>Communications</u>

- Three-channel, X-band, single-conversion repeater
- X-band beacon

<u>Tracking, Telemetry, and Command</u>

- S-band, compatible with SGLS
    - 165 commands
    - 215 telemetry points

<u>Antennas</u>

- Communications
    - One wide-beam receive (Northern Hemisphere)
    - One wide-beam transmit (Northern Hemisphere)
    - One narrow-beam transmit (European zone)
- Tracking, telemetry, and command
    - One omnidirectional ring array

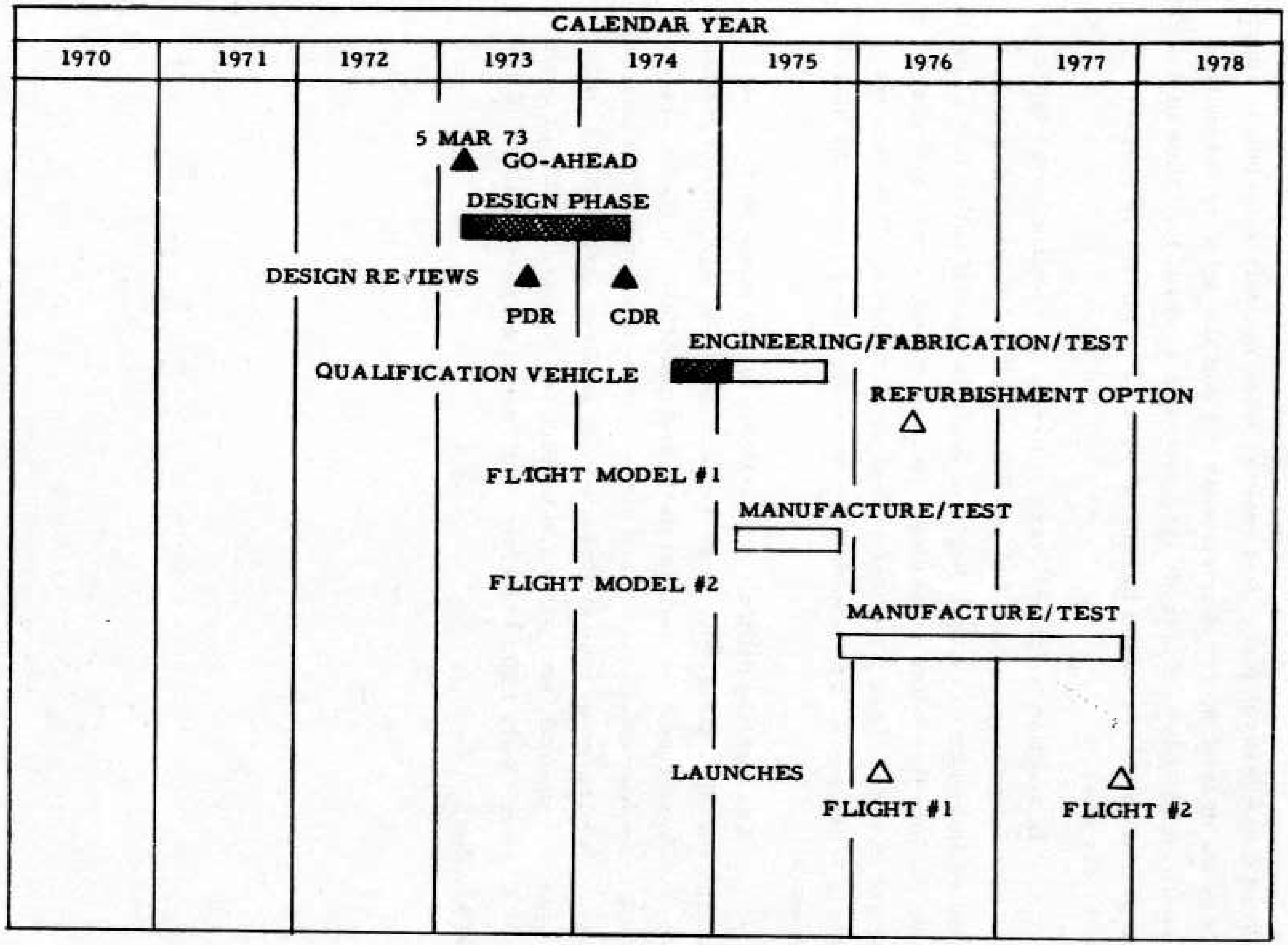



Figure C.14-2. Key Milestones and Events - NATO III

Philco-Ford found that the piece part vendors were not only reluctant to supply the parts to the special hi-rel requirements but that the delivery schedules were also stretched from an already long proposed average lead time of 33-36 weeks to periods of 42-52 weeks in some cases. The impact on program schedule is obvious.

In addition to late delivery of parts, 11 semiconductor types and two other critical part types failed to pass high-reliability screening tests. This resulted in vendors having to begin re-manufacture of entire lots of parts to replace the lots that failed screening. In a few cases, it was necessary for Philco-Ford to order these parts from other vendors because the vendors whose parts failed couldn't or wouldn't try to remake the parts to the hi-rel requirements.

The current difficulty in obtaining hi-rel piece parts is not unique to the NATO III program. A survey conducted by Air Force Systems Command Headquarters revealed that parts and materials shortages are impacting all weapon system, space, and other system development/acquisition programs. The degree of impact appears to be in direct proportion to the level of quality required from parts and materials vendors. NATO III quality standards are extremely high; therefore, the impact on parts deliveries is correspondingly great.

## C.15 INTELSAT I/II COMMUNICATIONS SATELLITES

### C.15.1 Intelsat I (Early Bird)

#### C.15.1.1 Program Summary

In August 1964 the International Telecommunications Satellite Consortium (now known as Intelsat) was organized. The purpose of Intelsat was the production, ownership, management, and use of a global communication satellite system. The feasibility of satellite communications had already been proven, and Intelsat decided to launch a satellite to gain information in four areas:

a. The rain margins required at ground stations
b. The reaction of telephone users to the transmission delay
c. The long-term operation of the stationkeeping control valves
d. The applicability of communication satellites for commercial telephone use

The satellite was to be basically experimental to provide some results in the above areas of uncertainty. If the results were favorable, the satellite would be put into operational use. Because of the success of Syncom, Intelsat decided to use a satellite of similar design. The resulting satellite (Intelsat I) was named Early Bird. It was designed and built by Hughes Aircraft Company of El Segundo, California, with Comsat Corporation acting as program manager for Intelsat.

Early Bird was launched in April 1965. Extensive tests were conducted using stations in Maine, England, France, and Germany, which had also operated with Telstar and Relay. Noise, intermodulation, and frequency response measurements were made with single and multiple carriers with voice and television signals. Optimal operating points for ground equipment were determined. The tests indicated that operation to commercial standards could be maintained. The DoD also conducted limited tests using Early Bird.

Early Bird was initiated into regular commercial service in June 1965 and operated regularly until January 1969. In July and August 1969 it was used again during a temporary outage of Intelsat IIIB.

C.15.1.2 **Satellite Description**

The Early Bird design basically followed the Syncom design. The bandwidth and radiated power were increased to provide better service, including two-way television. Larger solar cell panels were used, increasing the satellite height. Since the satellite was to be used for transmissions between North America and Europe, the antenna pattern was shaped to service the Northern Hemisphere. Maximum gain occurred at 45°N latitude, rather than at the equator. The satellite had two independent repeaters, one for transmissions from Europe to North America, and the other for the opposite direction. Details are given in Table C.15-1.

C.15.2 **Intelsat II**

C.15.2.1 **Program Summary**

Intelsat II was developed as a follow-on to Early Bird (Intelsat I). A prime factor in the timing of the Intelsat II program was NASA's need for multichannel communications with its overseas ground and shipborne tracking stations. Formerly these communication links depended on high-frequency radio, but the increase in manned space flights required improved quality and reliability. The Intelsat II satellites were designed to satisfy NASA's requirements and to have additional capacity for other commercial traffic. They were built by Hughes Aircraft Company of El Segundo, California.

The first Intelsat II satellite (IIA) was launched in October 1966, but because of an apogee motor malfunction, it's final orbit was elliptical with a synchronous altitude apogee. It was used for communications in the Pacific area a few hours a day until satellite IIB was launched. After that, it was used occasionally for ground station tests. Satellites IIB, IIC, and IID were all launched successfully and operated properly. They were used both for regular commercial service and in the NASA communications network. These three satellites, along with Early Bird, were "retired" in 1969 and 1970 when the Intelsat III satellites became operational. All four retired satellites are still in good condition and are available for further use when required.

Table C.15-1. Intelsat I (Early Bird) Technical Details

**Satellite**

- Cylinder, 28 in. diameter, 23 in. high
- 85 lb
- Solar cells, 45 W max. (NiCd batteries not used by the communication subsystem)
- Spin stabilized

**Configuration**

- Two 25-MHz-bandwidth double-conversion repeaters

**Capacity**

- 240 two-way voice circuits or 1 two-way TV circuit

**Transmitter**

- 4081 MHz to U.S., 4161 MHz to Europe
- Two TWTs, one on, one standby
- 6-W output, 10- to 11-dBW ERP per repeater

**Receiver**

- 6390 MHz from U.S., 6301 MHz from Europe
- 9-dB noise figure

**Antenna**

- Transmit - six-element colinear slot array, 9-dB gain, $11^\circ \times 360^\circ$ beam tilted $7^\circ$ above equatorial plane (maximum gain at about $45^\circ$N latitude)
- Receive - three-element cloverleaf array, 4-dB gain, $40^\circ \times 360^\circ$ beam

Table C.15-1. Intelsat I (Early Bird) Technical Details (Cont.)

Design Life

- 1.5 years

Orbit

- Synchronous equatorial

Launch Record

- Launched 6 April 1965
- Commercial service use from 28 June 1965 to January 1969, and from 29 June to 13 August 1969 (to fill coverage gap caused by Intelsat IIIB outage)

Developed By

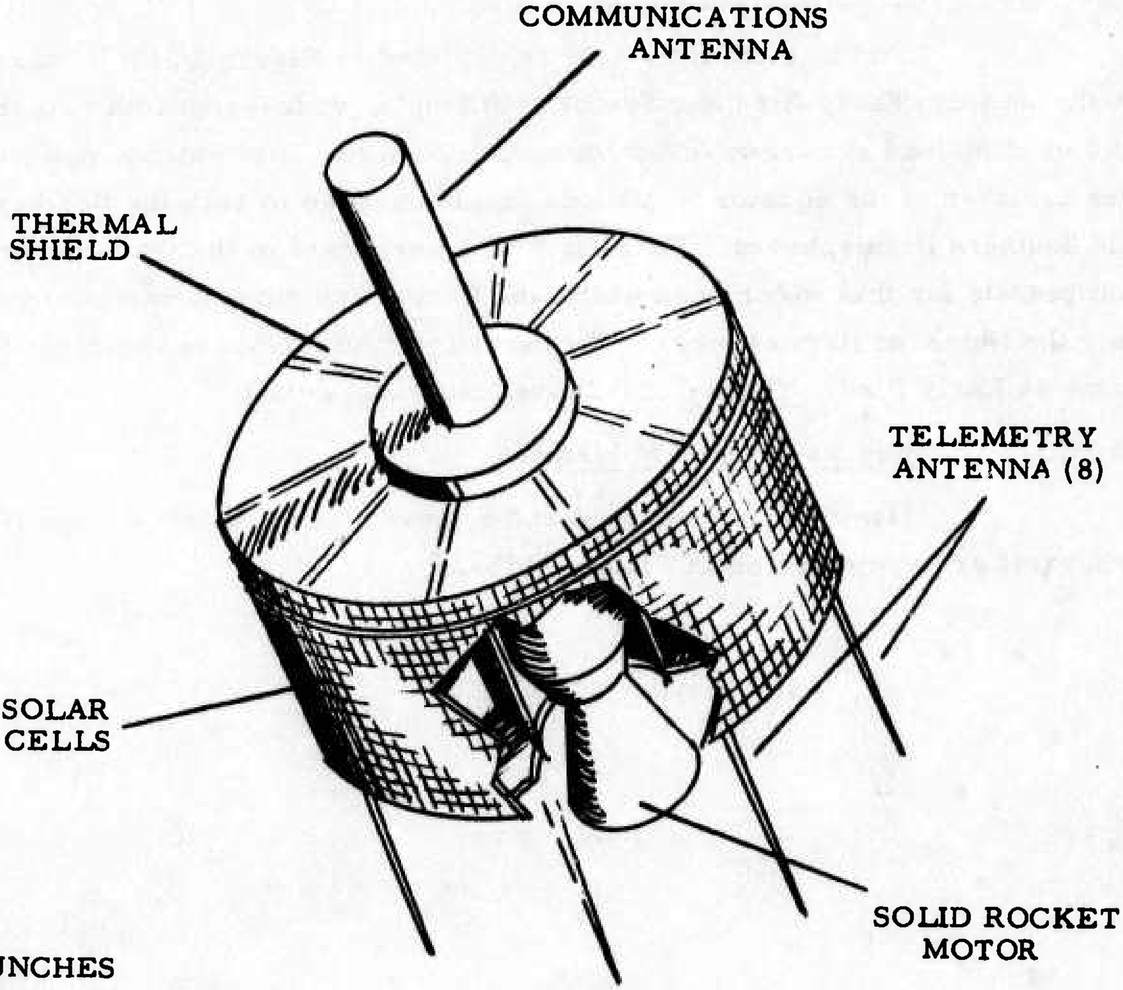
- Comsat Corporation
- Hughes Aircraft Company

C.15.2.2 Satellite Description

The Intelsat II design, as depicted in Figure C.15-1, was an evolution from Early Bird and Syncom. A single, wide-bandwidth repeater was used instead of narrow-band limiting repeaters. The antenna pattern was centered at the equator to provide equal coverage to both the Northern and Southern Hemispheres. Parallel TWTs were used in the transmitter to compensate for this wider beamwidth (the Early Bird antenna pattern covered only the Northern Hemisphere). The capacity of Intelsat II is therefore the same as Early Bird. Table C.15-2 gives technical details.

C.15.3 Key Events and Milestones

Significant milestones and problems in the Intelsat I and II programs are summarized in Figure C.15-2.




LAUNCHES

IIA (2F-1) - 26 OCT 1966
IIB (2F-2) - 11 JAN 1967
IIC (2F-3) - 7 APR 1967
IIC (2F-4) - 27 SEPT 1967

BOOSTER - THOR (TAT) DELTA

ORBIT - SYNC. EQUATORIAL

CHARACTERISTICS

| | |
|---|---|
| DIAMETER | - 56 IN |
| HT (OVERALL) | - 45 IN |
| WT (LIFTOFF) | - 357 LB |
| POWER (BOL) | - 85 WATTS |
| DESIGN LIFE | - 3 YEARS |

Figure C.15-1. International Telecommunications Satellite (Intelsat II)

Table C.15-2. Intelsat II Technical Details

Satellite

- Cylinder, 56 in. diameter, 26-1/2 in. high, overall height 45 in.
- 192 lb
- Solar cells and NiCd batteries, 85 W
- Spin stabilized

Configuration

- One 130-MHz-bandwidth single-conversion repeater

Capacity

- 240 two-way voice circuits

Transmitter

- 4055 to 4185 MHz
- Four 6-W TWTs; any combination of one, two, or three may be on
- 12-W output, 15.4-dBW ERP with two TWTs on

Receiver

- 6280 to 6410 MHz
- Redundant; one on, one standby
- 6-dB noise figure

Antenna

- Transmit - four-element biconical horn array, 5-dB gain
- Receive - single biconical horn, 4-dB gain

Table C. 15-2. Intelsat II Technical Details (Cont.)

Design Life

- 3 years

Orbit

- Synchronous equatorial

Launch Record

- IIA (F-1) - launched 26 October 1966, failed to achieve synchronous orbit (12-hr orbit allows 4 to 8 hr use per day, used in Pacific until IIB launched)
- IIB (F-2) - launched 11 January 1967, stationed over Pacific (retired mid-1969, available as spare)
- IIC (F-3) - launched 7 April 1967, stationed over Atlantic (retired February 1970, available as spare)
- IID (F-4) - launched 27 September 1967, stationed over Pacific (retired)

Developed By

- Comsat Corporation
- Hughes Aircraft Company

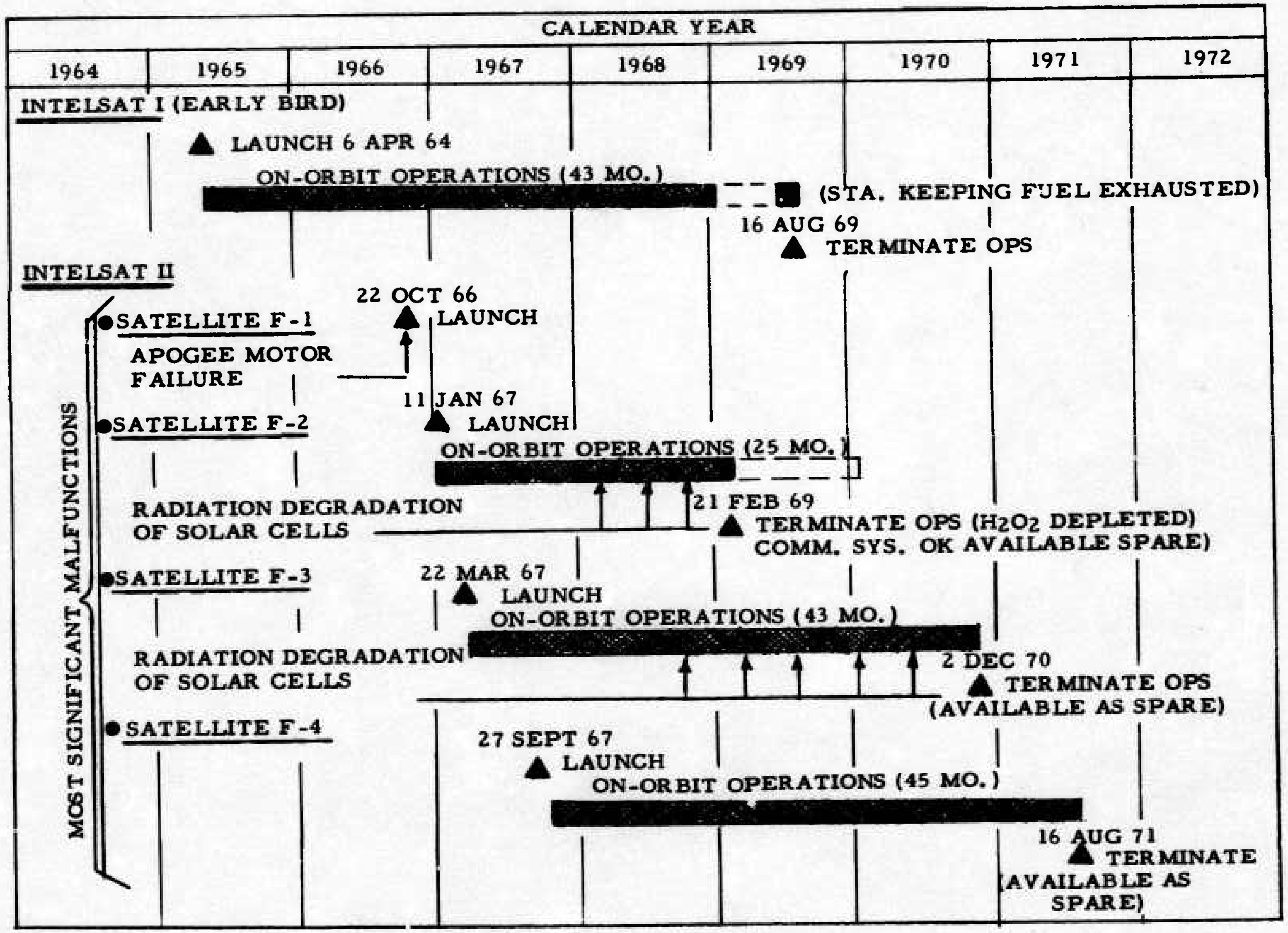



Figure C.15-2. Key Milestones and Events - Intelsat I and II

## C.16 INTELSAT III COMMUNICATIONS SATELLITE

### C.16.1 Program Summary

In 1966 work began on the Intelsat III satellites, which were to provide a significant increase in capacity over the previous satellites. A larger number of satellites was planned in order to provide full global coverage. The satellites were designed and built by TRW Systems Group of Redondo Beach, California, under the project management of Comsat Corporation.

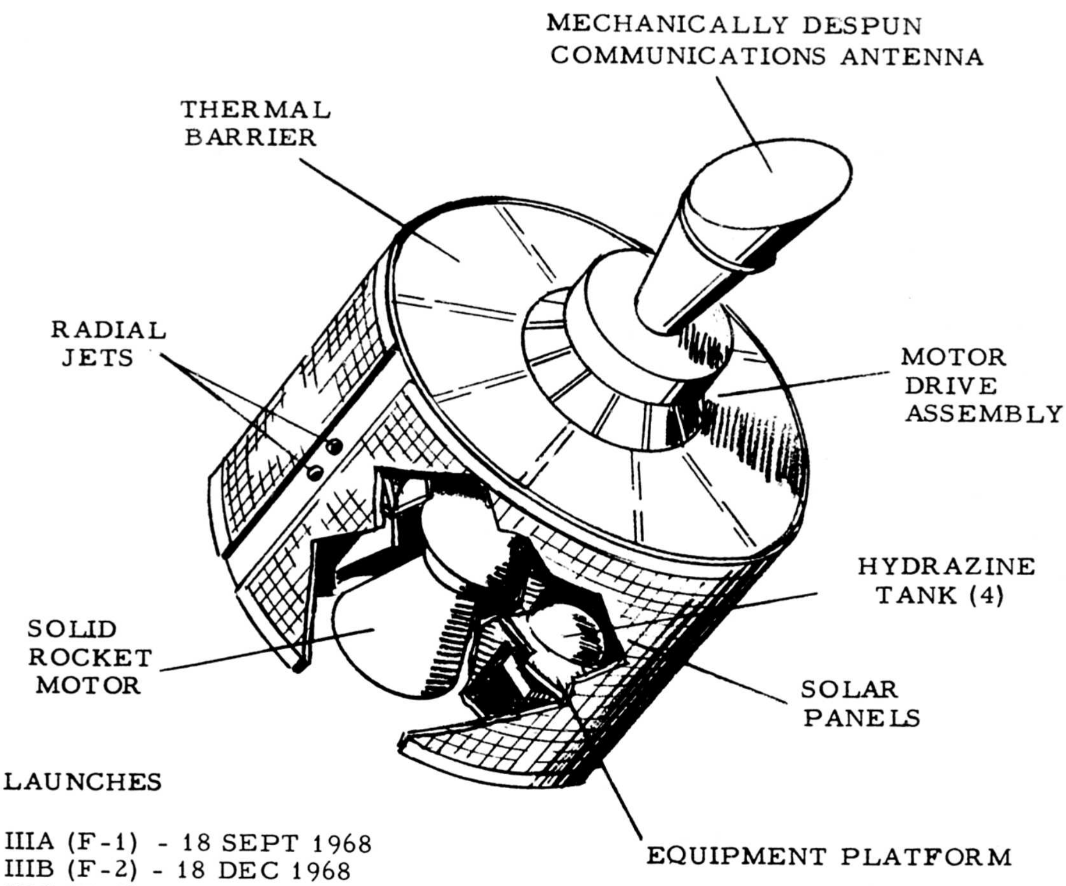
Originally the Intelsat III program was to include six launches. During the course of the program, partially due to the failure of the first launch, the program was extended to eight launches. The seventh satellite was fabricated from available spare parts, and the eighth was the refurbished prototype. Between December 1968 and July 1970, five of the eight satellites were successfully placed into synchronous orbit. All five operated satisfactorily. A component failure reduced the capacity of Intelsat IIIC, but it was moved from the Pacific to the Indian Ocean area, where it provided acceptable service due to the lower traffic density.

### C.16.2 Satellite Description

The size of Intelsat III was greater than Intelsat II because of increased payload capability of the Delta launch vehicles. In appearance (Figure C.16-1) it is a spinning cylinder similar to all the previous synchronous communication satellites. The major mechanical change is the use of a despun antenna with an earth-coverage beam. The increased gain available with the despun antenna is the major reason for the increase in capacity to 1200 voice circuits, from the 240 in Intelsat II and Early Bird. Details of the Intelsat III design are given in Table C.16-1.

### C.16.3 Key Milestones and Events

Significant milestones and problems in the Intelsat III program are summarized in Figure C.16-2. This information was obtained from References C.16-1 through C.16-5.




LAUNCHES

IIIA (F-1) - 18 SEPT 1968
IIIB (F-2) - 18 DEC 1968
IIIC (F-3) - 5 FEB 1969
IIID (F-4) - 21 MAY 1969
IIIE (F-5) - 25 JULY 1969
IIIF (F-6) - 14 JAN 1970
IIIG (F-7) - 22 APR 1970
IIIH (F-8) - 23 JULY 1970

BOOSTER - THORAD DELTA

ORBIT - SYNC. EQUATORIAL

CHARACTERISTICS

DIAMETER - 56 IN
HT (OVERALL) - 78 IN
WT (LIFTOFF) - 647 LB
POWER (BOL) - 160 WATTS
DESIGN LIFE - 5 YEARS

Figure C.16-1. International Telecommunications Satellite (Intelsat III)

Table C.16-1. Intelsat III Technical Details

Satellite

- Cylinder, 56 in. diameter, 41 in. high, overall height 78 in.
- ~ 330 lb
- Solar cells and NiCd batteries, 130 W
- Spin stabilized, 90 rpm

Configuration

- Two 225-MHz-bandwidth single-conversion repeaters

Capacity

- 1200 two-way voice circuits or 4 TV circuits

Transmitter

- 3705 to 3930 MHz and 3970 to 4195 MHz
- Each repeater has low-level TWT driving high-level TWT
- 10-W output, 27-dBW ERP each repeater (22 dBW minimum at edge of earth)

Receiver

- 5930 to 6155 MHz and 6195 to 6420 MHz
- Two tunnel diode amplifiers in each repeater (one on, one standby)
- < 7-dB noise figure

Table C.16-1. Intelsat III Technical Details (Cont.)

<u>Antenna</u>

- Mechanically despun conical horn with flat reflector 45° to horn axis
- 19.3° beamwidth, circular polarization
- Transmit - 18-dB peak gain
- Receive - 21-dB peak gain

<u>Design Life</u>

- 5 years

<u>Orbit</u>

- Synchronous equatorial

<u>Launch Record</u>

- IIIA(F-1) - launched 18 September 1968, failed to achieve proper orbit
- IIIB(F-2) - launched 18 December 1968, stationed over Atlantic
- IIIC(F-3) - launched 5 February 1969, stationed over Pacific (Indian Ocean after IIID launch)
- IIID(F-4) - launched 21 May 1969, stationed over Pacific
- IIIE(F-5) - launched 25 July 1969, failed to achieve proper orbit
- IIIF(F-6) - launched 14 January 1970, stationed over Atlantic
- IIIG(F-7) - launched 22 April 1970, stationed over Atlantic
- IIIH(F-8) - launched 23 July 1970, failed to achieve proper orbit

<u>Developed By</u>

- Comsat Corporation
- TRW Systems Group

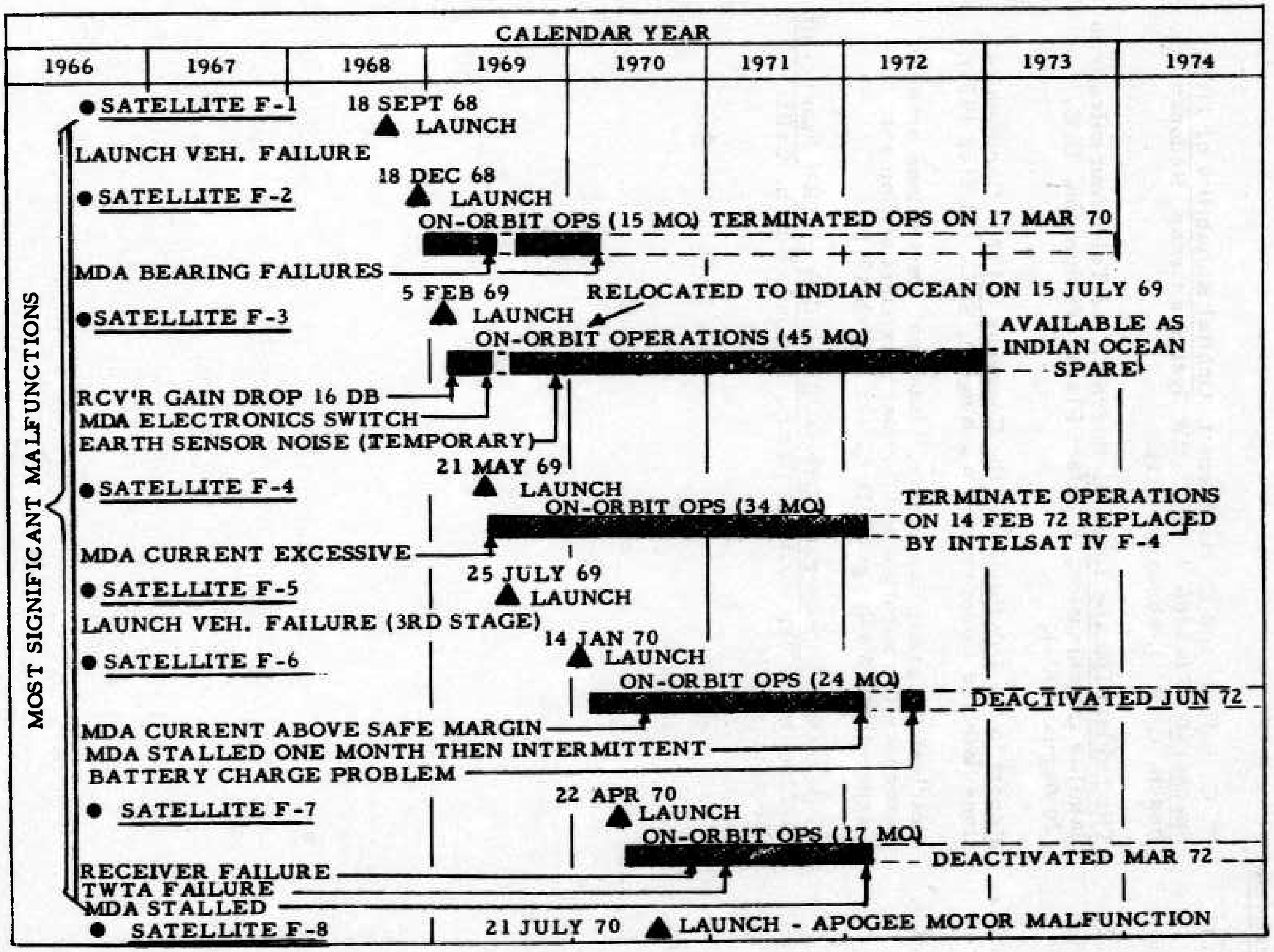



Figure C. 16-2. Key Milestones and Events - Intelsat III

References


C.16.1. R. C. Billups and E. H. Barnett, Orbital Reliability of TRW Spacecraft, 71-2286.1-019, TRW Systems Group, Redondo Beach, Calif. (February 1971).

C.16.2. The Intelsat Decase 1664-74, International Telecommunications Satellite Organization, L'Enfant Plaza, Washington, D.C. (20 August 1974).

C.16.3. Report to the President and the Congress - 1970, Communications Satellite Corporation, Washington, D.C. (31 May 1971).

C.16.4. "Intelsat Program Experience," informal discussions between Aerospace Corporation and Comsat personnel at Comsat Corporation, Washington, D.C. (25 March 1975).

C.16.5 E. Barnett, Demonstrated Orbital Reliability of TRW Spacecraft, 74-2286.142, TRW Systems Group, Redondo Beach, Calif. (December 1974).

## C. 17 INTELSAT IV COMMUNICATIONS SATELLITE

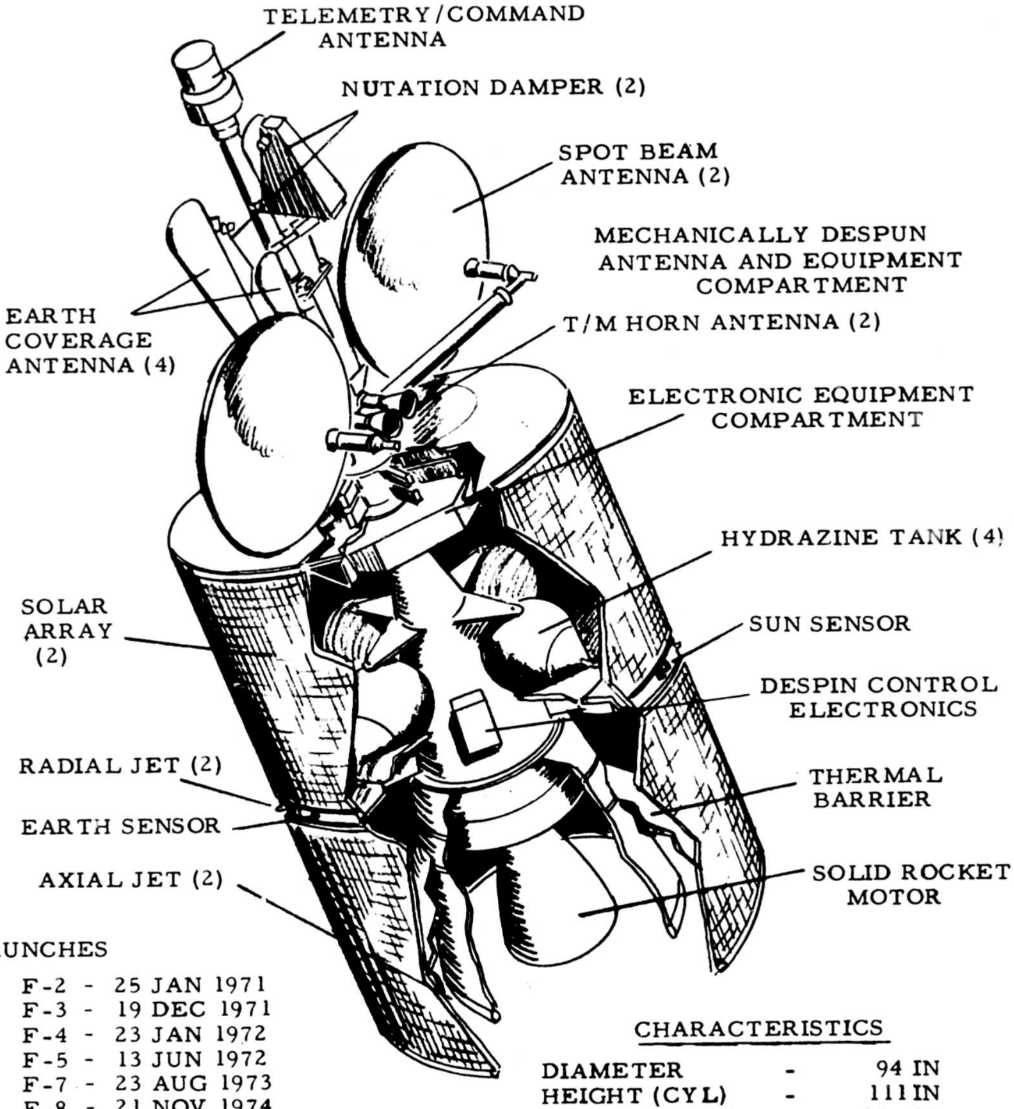
### C. 17. 1 Program Summary

Planning for the Intelsat IV satellites started before the first Intelsat III launch. The planning was based on the rather sure assumption that the demand for satellite communication services would increase yearly. The intent of Intelsat IV is to provide the required capacity increase; operationally it is compatible with the techniques and equipment used with Intelsat III. The Intelsat IV satellite is designed and built by the Hughes Aircraft Company of El Segundo, California, under the project management of Comsat Corporation.

Seven Intelsat IV satellites have been launched, six successfully; the first launch was in January 1971. The six that achieved orbit are now operational in the Intelsat network. As these satellites became operational, the smaller Intelsat III satellites were removed from active use and became on-orbit spares.

### C. 17. 2 Satellite Description

As shown in Figure C. 17-1, the design of Intelsat IV differs significantly from that of Intelsat III. The new design was necessary to achieve the required communication capacity. The basic satellite is similar to TACSAT, with a spinning shell and a despun platform on which are mounted the antennas and some electronics. Like TACSAT, and unlike earlier Intelsats, the spin axis is not the axis with the maximum moment of inertia, and appropriate attitude controls are required. Prior to Intelsat IV, increased capacity was obtained by increasing the repeater bandwidth, limited only by satellite ERP. Intelsat IV is a channelized system with 12 separate repeaters, each with 36 MHz bandwidth, to achieve efficient spectrum utilization. The total spectrum used is 432 MHz, and the capacity using earth-coverage antennas is 3000 duplex telephone circuits, or about 2-1/2 times the capacity of Intelsat III.




LAUNCHES

F-2 - 25 JAN 1971
F-3 - 19 DEC 1971
F-4 - 23 JAN 1972
F-5 - 13 JUN 1972
F-7 - 23 AUG 1973
F-8 - 21 NOV 1974
F-6 - 20 FEB 1975

BOOSTER - ATLAS/CENTAUR

ORBIT - SYNC. EQUATORIAL

CHARACTERISTICS

| | | |
|---|---|---|
| DIAMETER | - | 94 IN |
| HEIGHT (CYL) | - | 111 IN |
| HT (OVERALL) | - | 208 IN |
| WEIGHT (LAUNCH) | - | 3110 LB |
| POWER (BOL) | - | 569 WATTS |
| DESIGN LIFE | - | 7 YEARS |

Figure C.17-1. International Telecommunications Satellite (Intelsat IV)

Intelsat IV is the first satellite to have narrow-beamwidth (or spot beam) antennas in addition to earth-coverage antennas. These antennas have a 4.5° beamwidth (earth coverage is 17°) and are used only for transmitting. Up to four repeaters may be connected to each of the two narrow-beam antennas, providing a maximum capacity of 9000 telephone circuits. Additional details of the Intelsat IV design are given in Table C.17-1.

Ground equipment designed for use with Intelsat IV allows on-demand channel assignment as well as preassigned channels, which is the operating technique used with Intelsat I to III. For low-density communication links, on-demand assignment provides more efficient use of satellite capacity.

## C.17-3 Key Events and Milestones

Significant milestones of the Intelsat IV program and problems encountered during on-orbit operation of the satellites are summarized in Figure C.17-2.

Table C.17-1. Intelsat IV Technical Details

Satellite

- Cylinder, 94 in. diameter, 111 in. high body, overall height with antennas 210 in.
- 1559 lb
- Solar cells and NiCd batteries, 570 W initial, 460 W end of life
- Spin stabilized, 50 to 60 rpm, error $< \pm 0.35°$ (each axis)

Configuration

- Twelve 36-MHz-bandwidth single-conversion repeaters

Capacity

- Total of 3000 to 9000 two-way telephone circuits depending on use of earth-coverage or narrow-beam antennas, number of carriers per repeater, and modulation formats
- One color TV channel per repeater

Transmitter

- 3707 to 4193 MHz
- Two TWTs per repeater (one on, one standby)
- 6-W output per repeater
- ERP per repeater - 22.0 dBW (earth-coverage antenna), 33.7 dBW (narrow-beam antenna), both at -3 dB points of antenna pattern

Receiver

- 5932 to 6418 MHz
- Four complete units (one on, three standby), tunnel diode preamplifiers
- 8.2-dB noise figure

Table C. 17-1. Intelsat IV Technical Details (Cont.)

Antennas

- Four earth-coverage horns (two for transmit and two for receive), 20.5-dB gain peak, 17° beamwidth
- Two narrow-beam parabolas, 50-in. diameter, 31.7-dB gain peak, 4.5° beamwidth, steerable in the 17° earth-coverage cone
- All six antennas mounted on a despun platform and circularly polarized

Design Life

- 7 years

Orbit

- Synchronous equatorial

Launch Record

- F-2 - launched 25 January 1971, stationed over Atlantic (standby spare status)
- F-3 - launched 19 December 1971, stationed over Atlantic
- F-4 - launched 23 January 1972, stationed over Pacific
- F-5 - launched 13 June 1972, stationed over Indian Ocean
- F-7 - launched 23 August 1973, stationed over Atlantic
- F-8 - launched 21 November 1974, stationed over Pacific
- F-6 - launched 20 February 1975, failed to achieve proper orbit

Developed By

- Comsat Corporation
- Hughes Aircraft Company

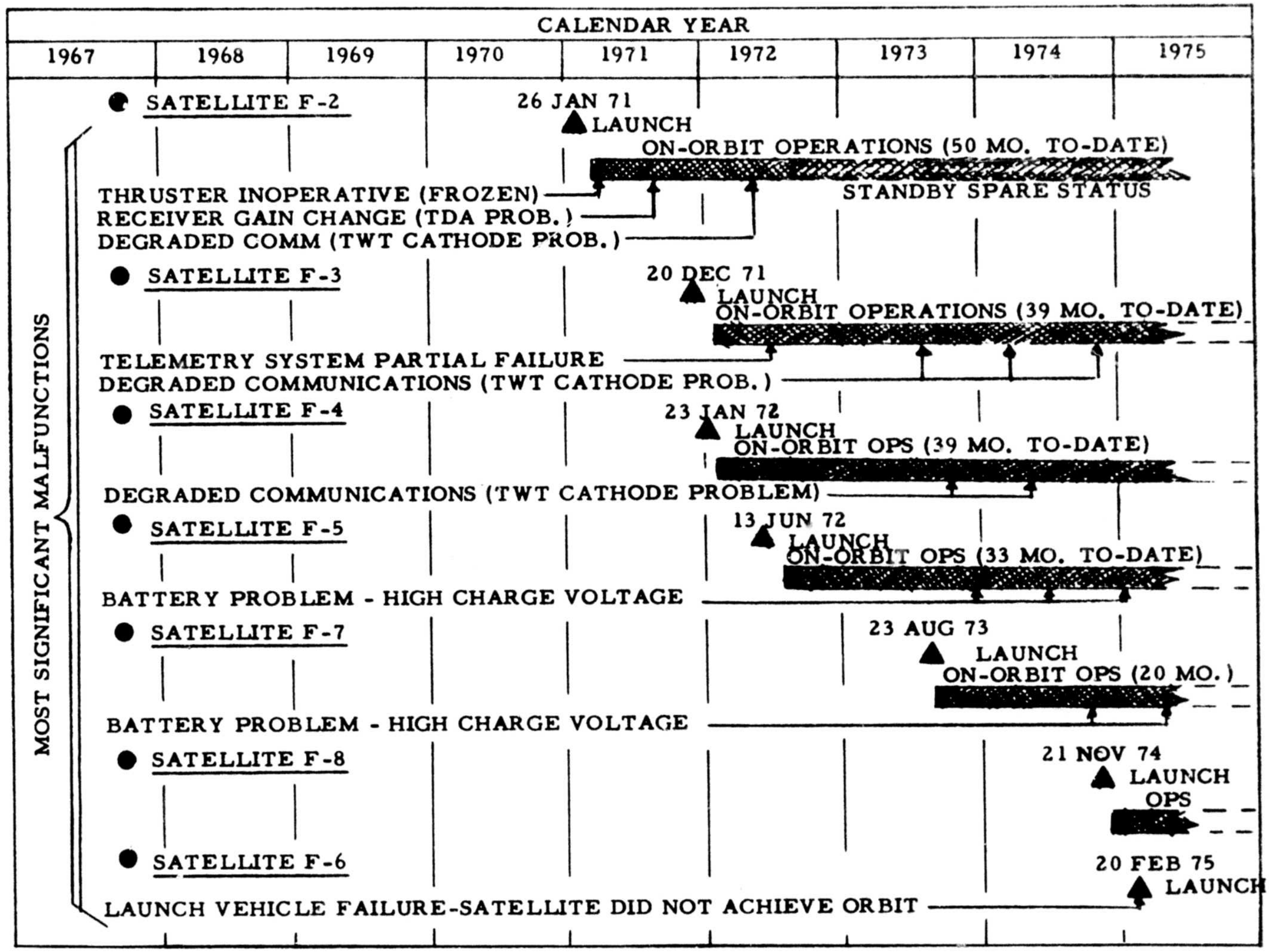



Figure C. 17-2. Key Milestones and Events - Intelsat IV

## C. 18 INTELSAT IVA COMMUNICATIONS SATELLITE

### C. 18. 1 Program Summary

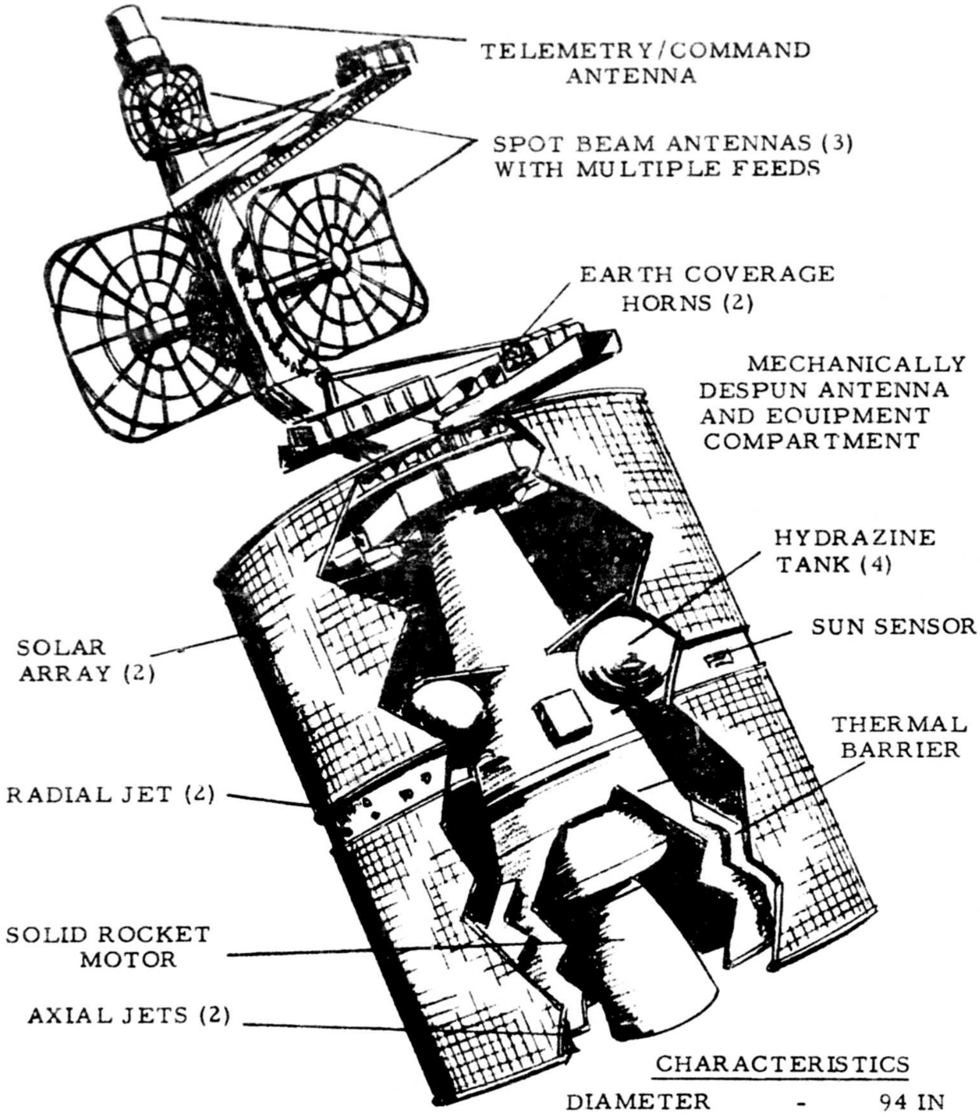
Currently, two Intelsat IV satellites are operating in the Atlantic area. According to the projections of capacity demand, these two satellites will be saturated by the end of 1975. To provide more capacity will require either a third Intelsat IV or a new satellite of larger capacity. Since the first alternative would force several ground stations to construct another antenna, Intelsat has chosen to develop Intelsat IVA. The displaced Intelsat IV satellite(s) will be moved to the Pacific or Indian Ocean areas, which have less traffic.

Three Intelsat IVA satellites are presently being assembled by Hughes Aircraft Company of El Segundo, California, under the project management of Comsat Corporation. The first will be launched in mid-1975 and will be positioned over the Atlantic Ocean. The second will be launched several months later and could be positioned over either the Atlantic or the Pacific Ocean. The third is currently designated as a spare. It is possible that additional satellites will be ordered to meet growing demands for capacity in all three ocean areas.

### C. 18. 2 Satellite Description

Many Intelsat IVA components are the same as Intelsat IV. The solar arrays are the same size on both satellites, but Intelsat IVA is taller because of the different antenna design. The new antennas and communication electronics allow an increase to twenty 36-MHz channels from the twelve on Intelsat IV. Intelsat IVA has five communication antennas: global coverage receive, global coverage transmit, spot beam receive, and two spot beam transmit. Figure C. 18-1 shows the Intelsat IVA satellite.

Four Intelsat IVA channels are devoted to global coverage. All four channels pass through one of the redundant global coverage receivers. Each channel has redundant 6-W TWTs. Sixteen channels are connected to the spot beam antennas. They are divided into an A group and a B group, each with eight channels. All the channels within a group use separate




CHARACTERISTICS

| | | |
|---|---|---|
| DIAMETER | - | 94 IN |
| HEIGHT (CYL) | - | 111 IN |
| HEIGHT (OVERALL) | - | 267 IN |
| WEIGHT (LAUNCH) | - | 3265 LB |
| POWER (BOL) | - | 590 WATTS |
| DESIGN LIFE | - | 7 YEARS |

LAUNCH DATE - MID 1975

BOOSTER - ATLAS/CENTAUR

ORBIT - SYNC. EQUATORIAL

Figure C.18-1. International Telecommunications Satellite (Intelsat IVA) frequenceis, but the passbands of one group of channels overlap those of the other group. There are four receivers for these channels, but only two are used at a time, one for each group. These channels use 5-W TWTs, with one spare TWT available for every two channels.

To prevent interference between overlapping channels, the spot beam antennas have east beams and west beams; one group of channels uses the east beams and the other group uses the west beams.* There is at least 27 dB isolation between the two sets of beams. The receive antenna has two sets of feed horns that produce the two east and west beams. One transmit antenna has four sets of feed horns that produce NE, NW, SW, and SE beams. The eastern pair of beams is isolated from the western pair, but the north and south members of a pair are not isolated since they carry no overlapping channels. Six channels are connected to the west beams and six to the east beams. Each of the channels connected to the east side may have its power split in any proportion between the NE and SE beams, and similarly for channels connected to the west side.

The other transmit antenna has two sets of feed horns, which produce a NE beam and a NW beam. Two channels are connected to each of the two beams. In an optional mode, two channels may be used for global coverage, in which case the other two must be turned off.

A considerable number of switches in the communication subsystem allows great flexibility in routing signals.

Each beam, on both the receive and the transmit antennas, is formed by a set of four or five feed horns, which shape the beam for coverage of the proper land masses. With only one exception, the coverage being used is adequate for Atlantic, Pacific, and Indian Ocean areas using fixed feed horns and fixed reflectors. This fact simplifies the satellites since no antenna gimballing is required, yet also allows the flexibility to move a

---

*The satellites are positioned over oceans, with the spot beams serving the continental areas on either side of the ocean. Any terminals near the satellite longitude are between the two beams and must use the global coverage channels.

satellite from one ocean area to another. The exception to the general coverage is a single additional feed horn that must be switched into the west receive beam and SW transmit beam to provide adequate coverage of New Zealand from a Pacific Intelsat IVA. Additional Intelsat IVA details are given in Table C.18-1.

## Table C.18-1. Intelsat IVA Technical Details

Satellite

- Cylinder, 94 in. diameter, 111 in. high, overall height 252 in.
- 1670 lb
- Solar cells and NiCd batteries, 590 W beginning of life, 490 W end of life
- Spin stabilized

Configuration

- Twenty 36-MHz-bandwidth single-conversion repeaters

Capacity

- Up to ~15,000 two-way telephone circuits, depending on number of carriers per repeater and modulation techniques
- One color TV channel per repeater

Transmitter

- 3707 to 4193 MHz
- Two 6-W TWTs (one on, one standby) for each of the four global coverage channels
- One 5-W TWT for each of the 16 spot beam channels, with one spare TWT for every two channels
- ERP per repeater at edge of coverage - 29 to 30 dBW (spot beam), 22 to 23 dBW (global)

Receiver

- 5932 to 6418 MHz
- Four receivers (two on, two standby)
- 8-dB noise figure, -11.6 dB/°K G/T[a]

Antennas

- Two earth coverage horns (1 transmit, 1 receive)
- Three spot beam antennas with multiple feeds to generate coverage patterns approximating continental shapes (2 transmit, 1 receive), at least 27-dB isolation between eastern and western lobes of each antenna
- All antennas mounted on a despun platform, and circularly polarized

Design Life

- 7 years

Orbit

- Synchronous equatorial, first satellite between 24° and 30°W longitude

Launch

- First launch mid 1975
- At least two launches
- Atlas/Centaur launch vehicle

Developed By

- Comsat Corporation
- Hughes Aircraft Company

[a]Gain/temperature ratio

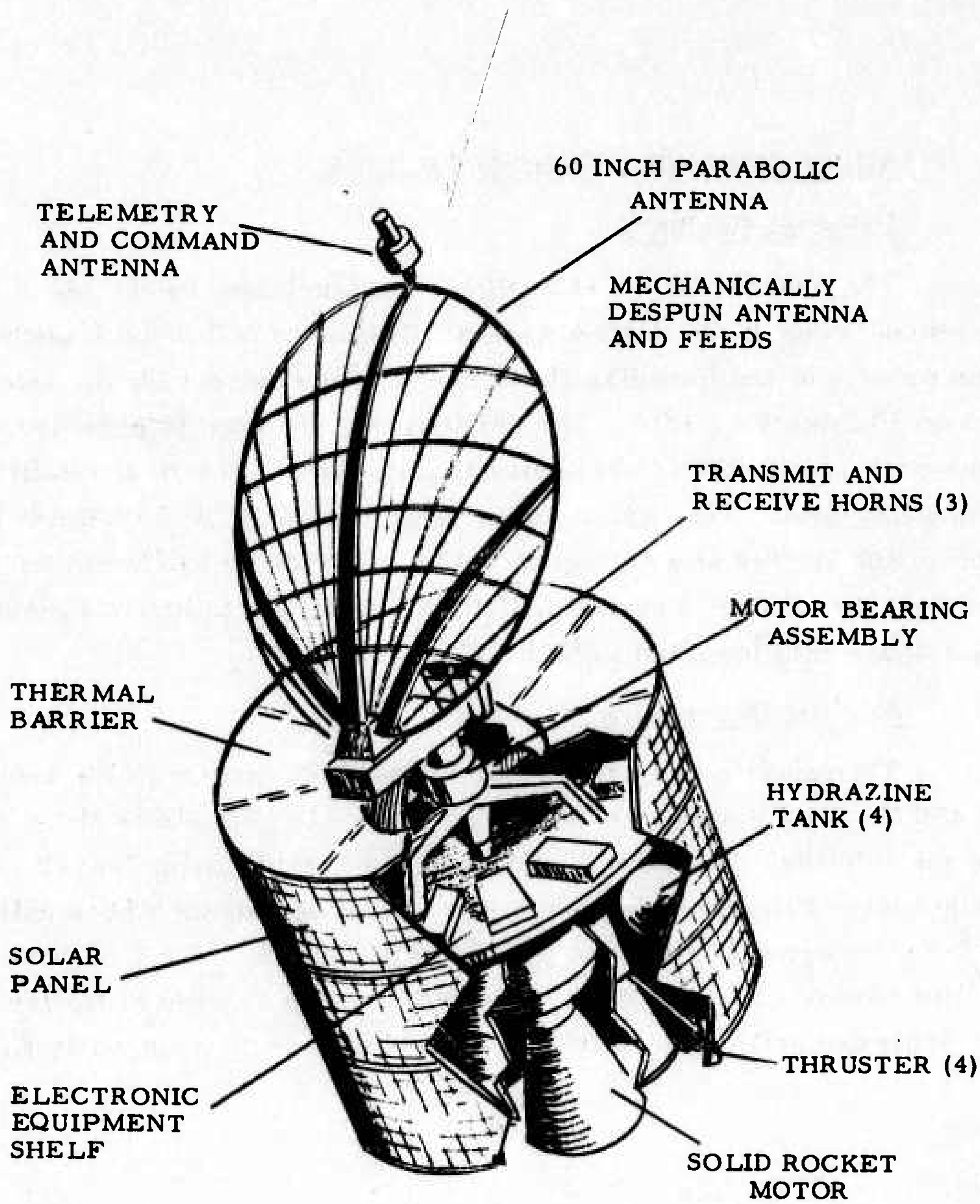
## C.19 ANIK COMMUNICATIONS SATELLITE

### C.19.1 Program Summary

The Anik satellite was designed and fabricated by the Space and Communications Group of the Hughes Aircraft Company for Telesat Canada, a corporation created by the Canadian Parliament. The contract for the satellite was signed on 30 September 1970. The satellite was designed to provide communications coverage for the entire Canadian land mass. The first satellite, launched on 9 November 1972, was successfully placed in orbit by a Delta 1914 launch vehicle and arrived at a station of 114°W longitude on 24 November 1972. The second Anik was launched on 20 April 1973. A third satellite was placed in storage as a spare until launched in May 1975.

### C.19.2 Satellite Description

The satellite is a cylindrical spinner (100 rpm) with the antenna at one end and the apogee motor at the other. Figure C.19-1 shows the primary features of the satellite. The Anik design is derived from the Intelsat IV satellite, the most significant differences being a single parabolic antenna whose pattern is optimized for coverage of Canada and the use of a single 5-W TWT in each of the 12 transmitter channels instead of the redundant 6-W TWTs used in Intelsat IV. The major items comprising the satellite subsystems are given in Table C.19-1.




**CHARACTERISTICS**

| | | |
|---|---|---|
| DIAMETER | - | 75 IN |
| HT (OVERALL) | - | 11.6 FT |
| WT (LIFTOFF) | - | 1200 LB |
| POWER (BOL) | - | 300 WATTS |
| DESIGN LIFE | - | 7 YEARS |

LAUNCHES

| | | |
|---|---|---|
| ANIK I | - | 9 NOV 1972 |
| ANIK II | - | 20 APRIL 1973 |
| ANIK III | - | 7 MAY 1975 |

BOOSTER - DELTA 1914

ORBIT - SYNC. EQUATORIAL

Figure C.19-1. Telesat Canada Communications Satellite (Anik)

## Table C.19-1. Anik Technical Details

**Satellites**

- Cylinder 75 in. diameter, 67 in. high, overall height 139 in.
- 650 lb
- Solar cells and NiCd batteries, 200 W end of life
- Spin stabilized, 100 rpm

**Configuration**

- Twelve 36-MHz-bandwidth single-conversion repeaters

**Capacity**

- 960 one-way voice circuits or one TV program per repeater

**Transmitter**

- 3702 to 4178 MHz
- One TWT per repeater
- 5-W output, 33-dBW minimum ERP over all of Canada

**Receiver**

- 5927 to 6403 MHz
- Two chains (one on, one standby) each with tunnel diode amplifiers and a low level TWT
- 7.8 dB noise figure

**Antenna**

- One offset feed parabola (50-in. diameter), linear polarization, beam shaped to maximize gain over Canadian territory, beam center tilted 7.85° north of equatorial plane

**Design Life**

- 7 years

**Orbit**

- Synchronous equatorial

**Launch Record**

- Anik I - 9 November 1972
- Anik II - 20 April 1973
- Anik III - May 1975

**Developed By**

- Telesat Canada
- Hughes Aircraft Company

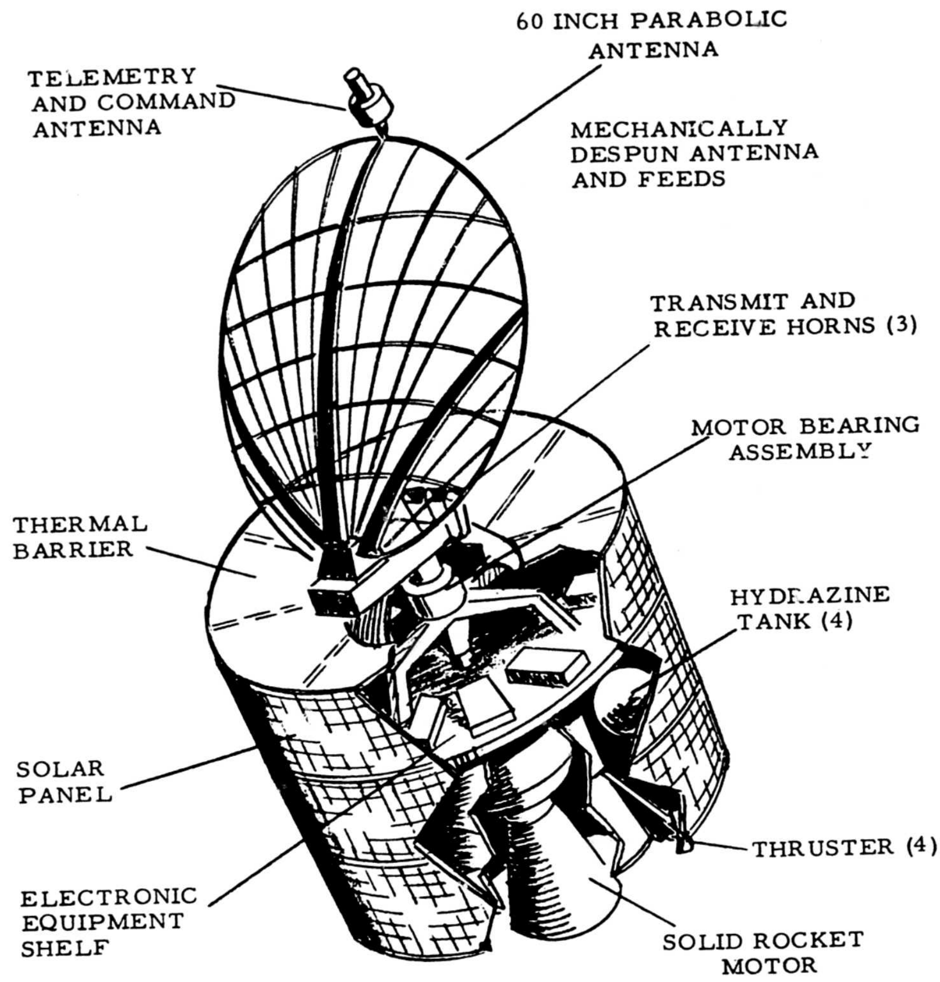
## C.20 WESTAR COMMUNICATIONS SATELLITE

### C.20.1 Program Summary

The WESTAR satellite is essentially identical to the Anik satellite, both of which were designed and fabricated by the Space and Communications Group of the Hughes Aircraft Company. The WESTAR was built for Western Union to provide commercial communications service for the continental U.S., Alaska, Hawaii, and Puerto Rico. The contract for satellite procurement was initiated in August of 1972. The first satellite was launched on 13 April 1974 and the second 10 October 1974 on a Delta 2914 launch vehicle. A third satellite was built as a spare. The on-orbit satellites are located at 91° and 99° W longitude.

### C.20.2 Satellite Description

The satellite is a cylindrical spinner (100 rpm) with the antenna at one end and the apogee motor at the other. Figure C.20-1 shows the primary features of the satellite. Antenna pointing, antenna feeds, and despin control electronics were the only substantive changes from the Anik satellite. The major items comprising the satellite subsystems are given in Table C.20-1.




CHARACTERISTICS

| | | |
|---|---|---|
| DIAMETER | - | 75 IN |
| HT (OVERALL) | - | 11.6 FT |
| WT (LIFTOFF) | - | 1233 LB |
| POWER (BOL) | - | 300 WATTS |
| DESIGN LIFE | - | 7 YEARS |

LAUNCHES

WESTAR I - 13 APRIL 1974
WESTAR II - 10 OCT. 1974

BOOSTER - DELTA 2914

ORBIT - SYNC. EQUATORIAL

Figure C.20-1. Western Union Communications Satellite (WESTAR)

Table C.20-1. WESTAR Technical Details

Satellite

- Cylinder, 75 in. diameter, 67 in. high, overall height 139 in.
- 650 lb
- Solar cells and NiCd batteries, ~300 W beginning of life, ~250 W end of life
- Spin stabilized, 100 rpm, $\pm 0.1°$ attitude control accuracy

Configuration

- Twelve 36-MHz-bandwidth single-conversion repeaters

Capacity

- Up to 1200 one-way voice circuits or one TV program per repeater

Transmitter

- 3702 to 4178 MHz
- One 5-W TWT per repeater (no TWT redundancy)
- ERP per repeater at edge of coverage - 33 dBW (CONUS), 27 dBW (Puerto Rico), 24.5 dBW (Alaska, Hawaii)

Receiver

- 5927 to 6403 MHz
- Two receivers (one on, one standby)
- -7 dB/°K (CONUS), -14 dB/°K (Alaska, Hawaii) G/T[a] at edge of coverage

Antenna

- One 60-in.-diameter reflector with three feed horns combined for coverage of CONUS and Puerto Rico and separate feed horns for Alaska and Hawaii, linear polarization

---

[a]Gain/temperature ratio

Table C.20-1. WESTAR Technical Details (Cont.)

Design Life

- 7 years

Orbit

- Synchronous equatorial

Launch Record

- WESTAR I - 13 April 1974
- WESTAR II - 10 October 1974
- Delta launch vehicle

Developed By

- Hughes Aircraft Company

## C.21 MARISAT MARITIME COMMUNICATIONS SATELLITE

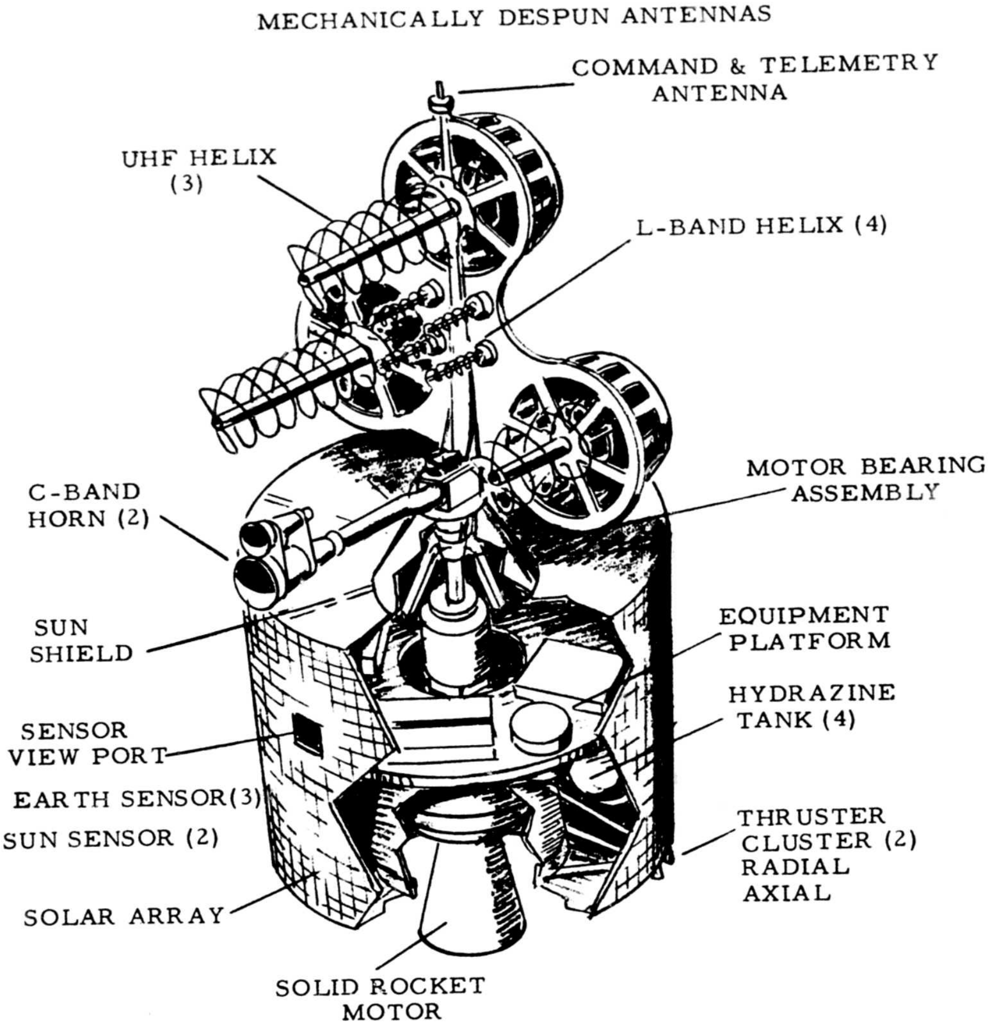
### C.21.1 Program Summary

MARISAT (also called GAPSAT) is a dual-mission synchronous orbit satellite intended to provide the U.S. Navy with a pre-FLTSATCOM satellite communications capability and also provide communications satellite service to commercial shipping. The MARISAT is a derivative of the Anik satellite and is built by the Hughes Aircraft Company for the Comsat General Corporation (and several other U.S. communication entities). Three satellites are being procured, two of which will form the on-orbit systems. The third is an on-the-ground spare. The satellites will be launched in 1975 on Delta 2914 launch vehicles with one satellite to be located over the Atlantic at 15°W longitude and the other over the Pacific at 176°E longitude. The antenna patterns provide horizon-to-horizon earth coverage.

### C.21.2 Satellite Description

The satellite is a cylindrical spinner with the antenna arrays at one end and the apogee motor at the other. Figure C.21-1 shows the primary features of the satellite. The design borrows heavily from the technology of Anik/WESTAR. Major changes are an increase in the solar panel diameter from 63 to 85 in. to provide more electrical power and a different communications subsystem (and antenna array) to fit the MARISAT mission. The major items comprising the satellite subsystems are given in Table C.21-1.

The MARISAT communications subsystem provides three UHF channels for Navy use, two with 25 kHz bandwidth and one with 500 kHz bandwidth. Each channel has a redundant transistor amplifier. For commerical use, there are two 4-MHz-bandwidth channels, one for ship-to-shore communications and one for shore-to-ship. These channels use L-band frequencies between the satellite and ships, and C-band between satellite and shore stations. TWTs are used for both L- and C-band




LAUNCH DATE
MID - 1975

BOOSTER - DELTA 2914

ORBIT - SYNC EQUATORIAL

CHARACTERISTICS

| | | |
|---|---|---|
| DIAMETER | - | 85 IN |
| HT (OVERALL) | - | 148 IN |
| WT (LIFTOFF) | - | 1445 LB |
| POWER (BOL) | - | 350 WATTS |
| DESIGN LIFE | - | 5 YEARS |

Figure C.21-1. Maritime Communications Satellite (MARISAT/GAPSAT)

Table C.21-1. MARISAT Technical Details

Satellite

- Cylinder, 85 in. diameter, 63 in. high, overall height ~ 12 ft
- ~ 720 lb
- Solar cells and NiCd batteries, 307 W end of life
- Spin stabilized

Configuration

- UHF - one 5-kHz channel and two 25-kHz channels
- L- and C-band - two 4-MHz channels (one L to C, one C to L)

Capacity

- 500-kHz channel - 8 voice circuits
- 25-kHz channels - 1 voice circuit each
- 4-MHz channel (L to C) - 8 voice circuits or > 100 teletype circuits
- 4-MHz channel (C to L) - 1, 4, or 8 voice circuits (depending on ERP) or > 100 teletype circuits

Transmitter

- UHF
    - ~ 250 MHz
    - Redundant solid state amplifiers
    - 66 W, 28 dBW (500-kHz channel), edge of earth
    - 21 W, 23 dBW (per 25-kHz channel), edge of earth

  L-Band
    - 1540 MHz
    - Redundant TWTs
    - 7, 29, or 64 W; 20, 26, or 29.5 dBW ERP, edge of earth

  C-band
    - 4197 MHz
    - Redundant TWTs
    - 5 W, 18.8 dBW ERP, edge of earch (if at saturation; however, this transmitter will always be operated linear)

Table C.21-1. MARISAT Technical Details (Cont.)

Receiver

- ~300 MHz, 1638.5 to 1642.5 MHz, 6420 to 6424 MHz
- Redundant receivers on each frequency
- G/T[a]
    - UHF - 18 dB/°K
    - L-band - 17 dB/°K
    - C-band - 25.4 dB/°K

Antennas

- UHF - 3 cone-helix antennas, 30° beamwidth
- L-band - 4 cone-helix antennas, ~20° beamwidth
- C-band - 2 horns (1 transmit, 1 receive), ~18° beamwidth

Design Life

- 5 years

Orbit

- Synchronous equatorial ( ≤ 3-1/2° inclination), 15°W and 176.5°E longitude

Launch Record

- Mid-1975
- Delta 2914 launch vehicle

Developed By

- Comsat General Corporation
- Hughes Aircraft Company

[a]Gain/temperature ratio at edge of earth

transmissions; the L-band TWT can be commanded to any of three power levels. The low power level at L-band will be used when all Navy channels are operating; when Navy requirements decrease or finish, the higher L-band power levels will be used.

MARISAT has nine communication antennas. Three helices backed by truncated cones form a UHF array with a 30° beamwidth. Four smaller cone-helix antennas form an L-band array with a 20° beamwidth. Two earth-coverage horns are used at C-band, one for transmitting and one for receiving.

## C.22 DOMESTIC COMMUNICATIONS SATELLITE (COMSTAR) - AT&T

### C.22.1 Program Summary

The COMSTAR satellites are part of a program by Comsat General Corporation, a subsidiary of Comsat Corporation, to provide domestic communications satellite service for AT&T. Under the program, four satellites are being constructed by the Hughes Aircraft Company. Delivery of the first flight spacecraft is scheduled for late 1975. Comsat General will own and operate these satellites and lease their communications capacity to AT&T under the terms of a 7-year agreement (Ref. C.22-1).

The total system will consist of three Comsat General satellites in geosynchronous orbit; one ground spare; two Comsat General TT&C earth stations; five AT&T communications earth stations; and a Comsat General system control center. The ground spare will be available in case of an unsuccessful launch or an on-orbit failure. The launch vehicle will be an Atlas/Centaur, with the first launch scheduled for late 1975.

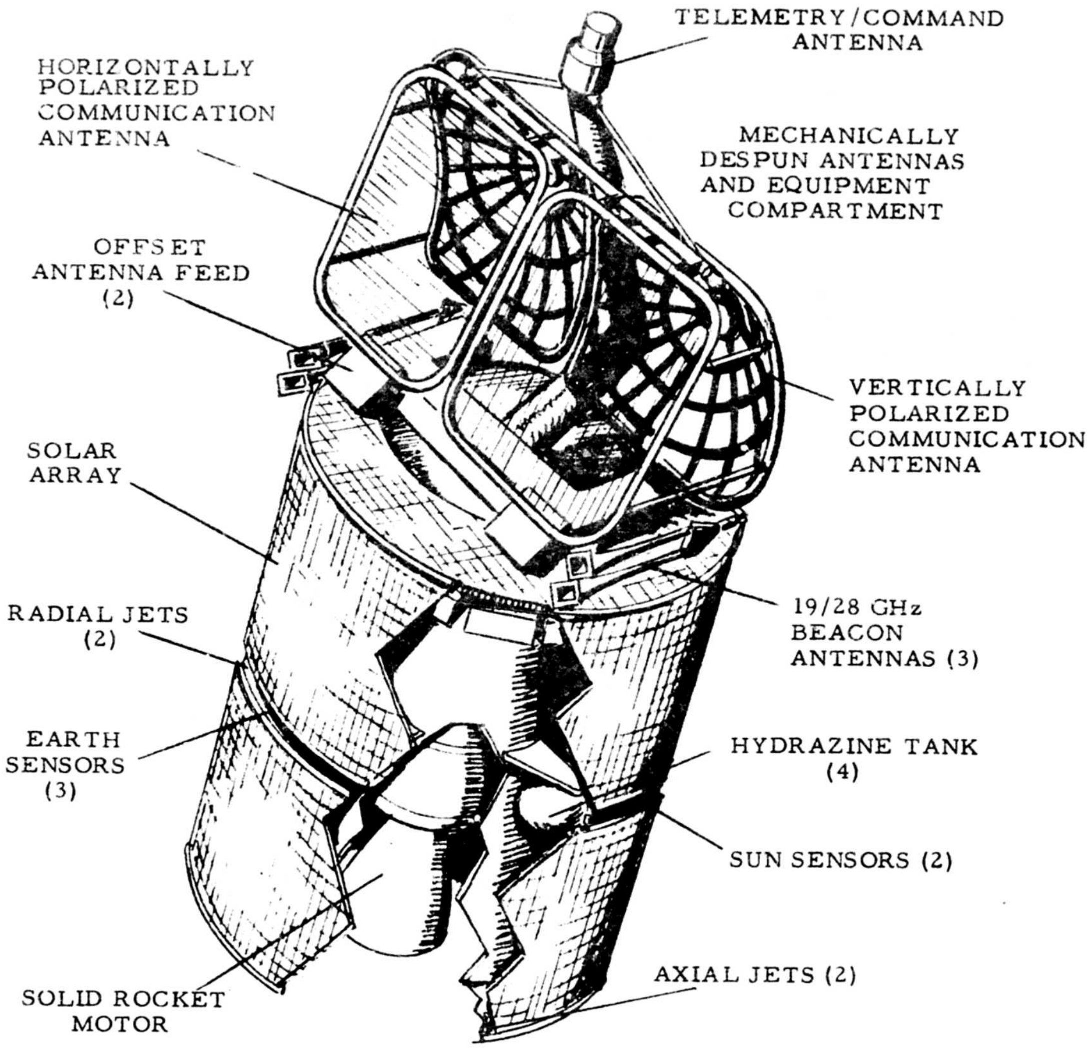
### C.22.2 Satellite Description

Overall, the cylindrical satellite will be about 8 ft in diameter, have a height of 20 ft, and weigh about 3255 lb at launch. Its configuration and characteristics are shown in Figure C.22-1. A technical summary is given in Table C.22-1.

#### C.22.2.1 General Features (Ref. C.22-1)

The satellite utilizes a dual-spin design with the spin axis perpendicular to the orbital plane. The two main elements of the spacecraft are the spinning rotor and the despun earth-oriented platform containing the communications repeater and its antennas. The spinning rotor provides the basic gyroscopic stability to the spacecraft.

The reaction control subsystem is mounted on the rotor. Redundant jets are provided for orbital stationkeeping, attitude control, and spin-up with sufficient fuel (hydrazine) for the 7-year lifetime of the satellite.




LAUNCH DATE - LATE 1975

BOOSTER - ATLAS/CENTAUR

ORBIT - SYNC. EQUATORIAL

CHARACTERISITICS

| | | |
|---|---|---|
| DIAMETER | - | 94 IN |
| HEIGHT (CYL) | - | 111 IN |
| HEIGHT (OVERALL) | - | 240 IN |
| WEIGHT (LIFTOFF) | - | 3255 LB |
| POWER (BOL) | - | 600 WATTS |
| DESIGN LIFE | - | 7 YEARS |

Figure C.22-1. Domestic Communications Satellite Program (COMSTAR) - AT&T

Table C.22-1. COMSTAR Technical Details

Satellite

- Cylindrical, 94 in. diameter, 111 in. high, overall height 240 in. including antennas
- 3255 lb
- Solar cells and NiCd batteries (2) - 550 W after 7 yr
- Spin stabilized

Configuration

- 24 transponders - 34-MHz bandwidth

Capacity

- More than 28,000 simultaneous telephone conversations or various combinations FDM, TV, and TDM signals

Transmit

- 3700 to 4200 MHz
- 33.0 dBW EIRP per transponder at beam edge (31 dBW for CONUS/Alaska combined)
- Transponder distribution
    - 6 to 24 CONUS
    - Up to 6 assigned to Alaska or CONUS/Alaska combined
    - Up to 6 assigned to Hawaii
    - Up to 6 assigned to Puerto Rico
- 19/28 GHz beacon package

Receive

- 5925 to 6425 MHz
- Four receivers (two on, two standby)
- -9 dB/°K G/T[a] at beam edge (at 6 GHz)

[a]G/T - Gain/temperature ratio.

Table C.22-1. COMSTAR Technical Details (Cont.)

- Normal transponder distribution
  - 12 CONUS and Alaska
  - 12 CONUS, Hawaii, and Puerto Rico

Antennas

- Two gridded parabolic, offset fed antennas
  - One vertically polarized
  - One horizontally polarized
- 33 dB polarization isolation
- One telemetry/command antenna
- 19/28 GHz beacon antennas
- All antennas on despun platform

Design Life

- 7 years

Orbit

- Synchronous equatorial

Launch

- At least three launches
- First launch late 1975
- Atlas/Centaur launch vehicle

Developed by

- Comsat Corporation
- Hughes Aircraft Company

The directional communication antennas are pointed to the proper point on the earth by the despin control system. Redundant sensor information (three earth sensors, two sun sensors, all rotor-mounted) is used by an onboard processor to establish the inertial attitude of the spacecraft and control the antenna platform.

The apogee motor is carried in the aft half of the rotor section. This motor provides the needed impulse to place the spacecraft in its final synchronous orbit.

The spacecraft temperature is controlled passively by selecting in the design phase the proper ratio of the solar energy absorptivity of the various external surfaces to their infrared emissivity. Active heaters are also provided on certain temperature-sensitive elements of the satellite equipment.

The power subsystem is located on the spinning rotor; included are two cylindrical solar arrays and two nickel cadmium batteries. The capacity of the batteries is sufficient to provide continuous service of all 24 transponders during eclipse seasons throughout the 7-year life span.

The despun section of the satellite contains the communications repeater electronics and telemetry and command electronics. Spacecraft commanding is performed through two cross-strapped command systems. Telemetry information from the spacecraft is provided by redundant pulse-code-modulated (PCM) systems with a frequency-modulated (FM) real-time mode available for transmission of certain data during spacecraft maneuvers.

### C.22.2.2 Communications Subsystem (Ref. C.22-1)

Each satellite will contain 24 transponders, all of which can serve the continental United States (CONUS). Six of the transponders can be switched either to cover Alaska only or a combined CONUS/Alaska coverage. In the CONUS/Alaska mode, the satellite's transmitter power is directed to both the 48 contiguous states and to Alaska. In addition, six other transponders can be switched from CONUS to Hawaii and another six to Puerto Rico.

The receiving portion of the communications subsystem consists of two primary receivers with two backup receivers, each capable of replacing either of the primary receivers by ground command. One primary receiver can be accessed by 12 channels from CONUS and Alaska, and the other primary receiver can be accessed by 12 channels from CONUS, Hawaii, and Puerto Rico in the uplink 6-GHz frequency band and at their specific polarizations. Each receiver amplifies the 12 channels and converts them to 4 GHz. The output of each receiver then passes to a 12-channel input multiplexer where each channel is separately filtered, amplified, and then fed to the output multiplexer where the channels are combined in banks of six and fed to the transmit antennas.

To provide frequency re-use within the same satellite, one set of 12 channels is cross-polarized and frequency-staggered by 20 MHz to the other 12 frequency-sharing transponders. The isolation provided by the spacecraft antenna between the two polarizations is greater than 33 dB over the coverage areas.

The antenna subsystem consists of two gridded, offset fed parabolic transmit and receive antennas, one horizontally polarized and the other vertically polarized. The feeds are properly offset to obtain the coverage previously described. The antennas are aimed at the proper point on the earth by the despin control system.

C.22.2.3 <u>19/28-GHz Beacons (Ref. C.22-1)</u>

The satellites will also contain a 19/28-GHz beacon package to provide SHF signal sources for use in gathering data on space-to-earth signal propagation. Attenuation, depolarization, and phase coherence will be measured at 19.0 and 28.6 GHz. The data will help to determine minimum power and other performance margins needed for future satellite communications systems operating above 10 GHz.

The transmitters to be used in the propagation experiments will employ an all-solid state design using newly developed IMPATT amplifiers for the microwave power amplification.

A stable master oscillator will allow the signals to be measured by a receiver with a bandwidth of 100 Hz at 19 GHz and 150 Hz at 28.6 GHz. Precise passive temperature control and careful circuit design will maintain the power output constant within approximately 0.5 dB.

The 19-GHz beacon will provide horizontally and vertically polarized transmissions alternated at a 1000-Hz rate with an EIRP greater than 21 dBW over CONUS. The 28-GHz beacon will provide approximately the same EIRP over the CONUS with a vertically polarized transmission having two sidebands spaced 264/528 MHz from the carrier. These high EIRPs will allow accurate measurement of propagation effects by earth stations having small inexpensive antennas.

Reference

C. 22-1. Domestic Communications Satellites, brochure, Comsat General Corporation, Washington, D.C. (December 1974).

## C.23 JAPANESE MEDIUM-CAPACITY COMMUNICATIONS SATELLITE (CS)

### C.23.1 Program Summary

The CS medium-capacity communications satellite is being built by Philco-Ford Corporation, at its Western Development Laboratories Division (WDL) in Palo Alto, California. It will be used by the Japanese Government or its agencies for communications experiments with satellite links between fixed and mobile ground stations covering the Japanese Islands.

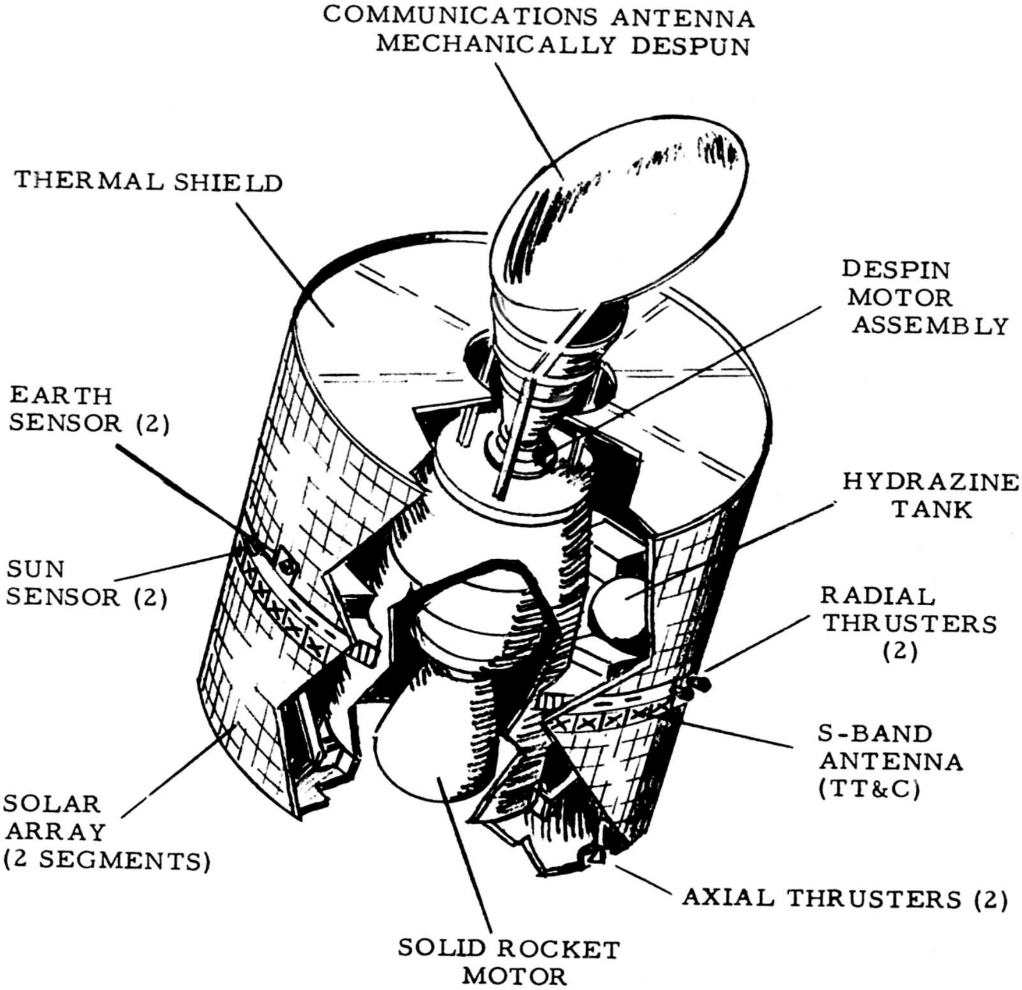
The initial launch is scheduled for 1977 from the Eastern Test Range. The launch vehicle will be a Delta 2914.

### C.23.2 Satellite Description

The CS satellite will be a spin-stabilized spacecraft, with mechanical despin motors used to despin only the communications antenna. The general characteristics of the satellite are shown in Figure C.23-1. Key features are:

- The design is simple and based on previously flight proven concepts and hardware.
- The satellite is spin-stabilized in orbit.
- The satellite is inherently stable through the mission, including transfer orbit.
- Maximum use of existing equipment and techniques is supplemented by thorough breadboarding and testing of new payload equipment.
- All communication performance requirements are met within the capabilities of the TWT.
- The high gain, multifrequency antenna is fixed and does not require nodding.
- All mechanisms are fixed and do not require deployment.
- In the event of battery failure, the satellite will function in sunlight without special commanding from the ground.
- Attitude control, antenna pointing, and stationkeeping are ground commandable.

JAPANESE SAT - PHILCO FORD




LAUNCH DATE

LATE 1977

BOOSTER - DELTA 2914

ORBIT - SYNC. EQUATORIAL

CHARACTERISTICS

| | | |
|---|---|---|
| DIAMETER | - | 86 IN |
| HT (OVERALL) | - | 139 IN |
| WT (LIFTOFF) | - | 1490 LB |
| POWER (BOL) | - | 538 WATTS |
| DESIGN LIFE | - | 3 1/2 YEARS |

Figure C.23-1. Medium Capacity Communications Satellite (CS)

- The system and subsystem design preclude physical damage to the satellite and subsystems from inadvertent commands.
- The satellite can be launched any day of the year.

The satellite subsystems are summarized in Table C.23-1.

## Table C.23-1. CS Satellite Technical Details

Configuration and Structure

- Cylindrical, 86 in. diameter, 88 in. long, overall length 138.7 in. including communication antenna
- 1490 lb at launch
- Entire spacecraft spinning with communication antennas despun
- Monocoque shell supporting single horizontal equipment platform

Stabilization

- Spin-stabilized (90 rpm)
- Stable configuration in transfer orbit and throughout on-station life

Propulsion

- Aerojet SVM-6 solid propellant apogee kick motor

Reaction Control

- Redundant 5-lb thrusters (hydrazine monopropellant)
- Attitude, velocity control
- N-S stationkeeping

Thermal Control

- Passive except for localized heaters
- Heat transfer mainly through ends of satellite

Attitude and Antenna Control

- Earth sensors (2) and sun sensors (2) for attitude reference
- Antenna despin motor with torque margin and independent, redundant windings
- Redundant electronics and nutation damper

Electrical Power

- Two-segment cylindrical solar array
  - BOL - 538 W (autumnal equinox), 468 W (summer solstice)
  - EOL - 421 W (autumnal equinox), 375 W (summer solstice)
- One 20-cell NiCd battery, 20 A-hr
- Separate battery charge array
- Partial shunt-regulated main bus (29.4 ± 0.2 Vdc bus reg.)
- Automatic load control for turn-on sequence

Table C.23-1. CS Satellite Technical Details (Cont.)

Communications

- One despun antenna (circular omni) with multifrequency feed system
- Two C-band (6/4 GHz) channels
  - Single-conversion receiver
  - TDA preamplifiers
  - 200-MHz channel bandwidth
  - 34.5-dBm output/channel
  - 9-dB input noise figure
- Six K-band (29/19 GHz) channels
  - No preamplifier
  - 200-MHz channel bandwidth
  - 34-dBm output/channel
  - 13-dB input noise figure
- Both C-band and K-band tracking beacons
- Capable of 100 Mbps transmission at C-band

Telemetry, Tracking, and Command

- STDN compatible
- Ring array, near-isotropic coverage
  - 21.5-dBm EIRP
  - 9-dB uplink margin
  - 128 commands
  - 128 8-bit TLM words

Design Life

- Goal - 3 years
- Expendables sized for 3.5 years

Launch Date

- Late 1977 (from ETR)

## C.24 RCA SATCOM

### C.24.1 Program Summary

The RCA SATCOM was designed and developed by the RCA Astro-Electronics Division in Princeton, New Jersey. The procuring agencies are RCA Global Communications, Inc. and RCA Alaska Communications, Inc.

The contract start date was 23 October 1973. Three flight models are under construction, the first to be launched in December of 1975 on a Delta 3914 booster. The satellites will be deployed in synchronous equatorial orbits to service the continental U.S., Hawaii, and Alaska.

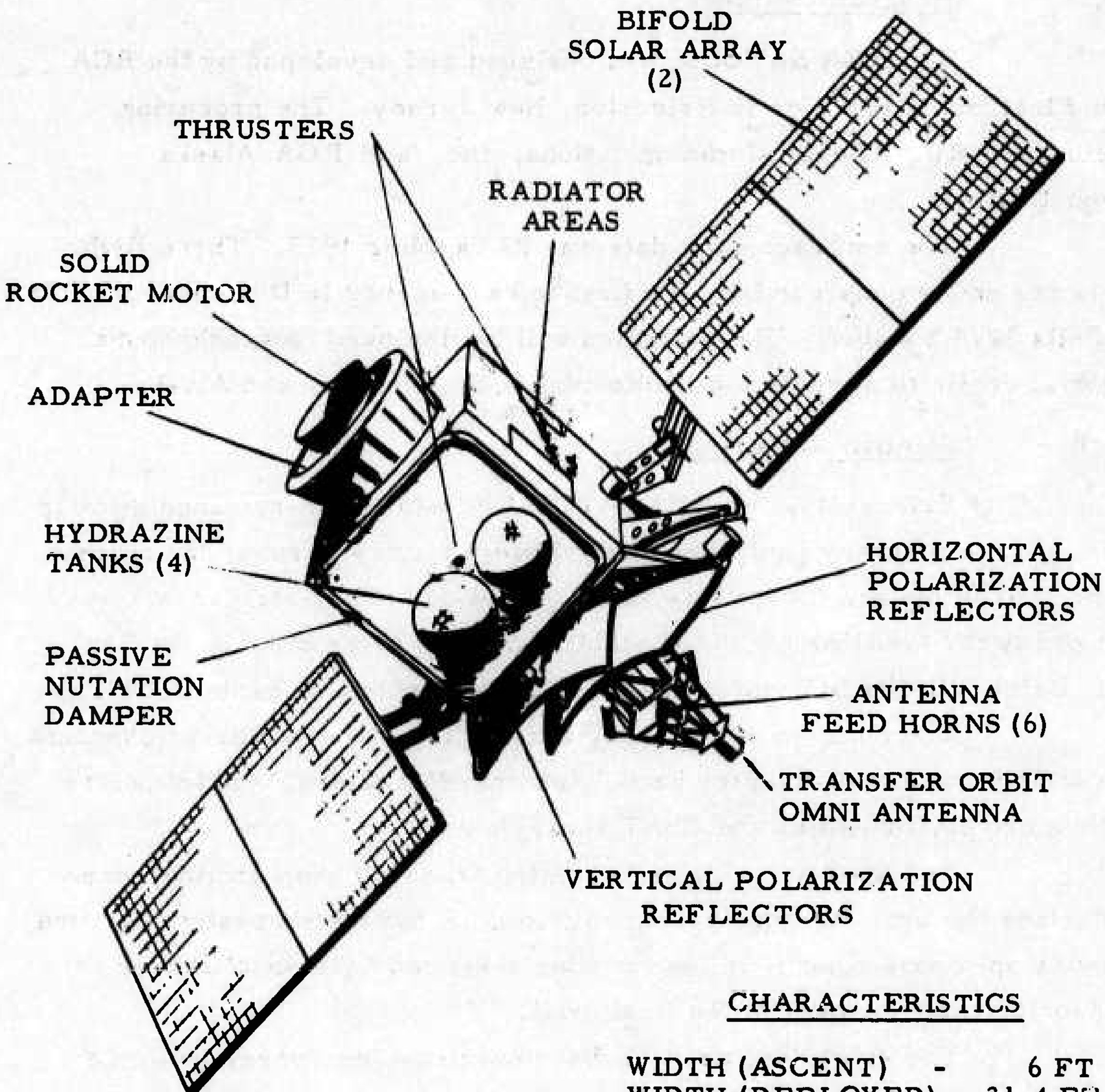
### C.24.2 Satellite Description

Externally, the RCA SATCOM consists of a rectangular body that houses the primary equipment, two deployed solar arrays, the antenna system, and an apogee motor (see Figure C.24-1). The satellite is spin-stabilized in the transfer orbit and stabilized about three axes in the final orbit. Brief satellite subsystem descriptions are given in Table C.24-1.

The communications payload employs 24 interleaved channels in the 6/4 GHz common-carrier band. Command, ranging, and telemetry functions are performed by the CR&T subsystem.

The apogee motor removes the transfer orbit inclination and circularizes the orbit at synchronous altitude. A hydrazine system is fitted to provide spin-axis control in the transfer orbit and both east/west and north/south stationkeeping in the final orbit.

The electrical power subsystem provides energy from the arrays during sunlight operation and battery power for full satellite operation for eclipse operation. A heater-augmented passive thermal control subsystem maintains components within operating temperatures.




**CHARACTERISTICS**

| | | |
|---|---|---|
| WIDTH (ASCENT) | - | 6 FT |
| WIDTH (DEPLOYED) | - | 31.4 FT |
| HEIGHT (OVERALL) | - | 4.5 FT |
| WT (LIFTOFF) | - | 1925 LB |
| POWER (BOL) | - | 740 WATTS |
| DESIGN LIFE | - | 8 YEARS |

| | | |
|---|---|---|
| LAUNCH DATE | - | DEC 1975 |
| BOOSTER | - | DELTA 3914 |
| ORBIT | - | SYNC EQUATORIAL |

Figure C.24-1. Global Communications Satellite (RCA SATCOM)

Table C.24-1. RCA SatCom Subsystem Description

General

- Rectangular body with deployed solar arrays
- 47 × 64 × 44 in. body
- 1960 lb at launch, 770 lb on-orbit
- 8-year reliability (20 of 24 channels) @ 0.5 probability of success
- Apogee boost motor for final orbit insertion

Structure

- Central cylinder
- Rectangular panels
- Cylinder to panel webs
- Solar array support structure

Attitude Control

- Spin stabilization in transfer orbit
- Electronics
- Single body-mounted momentum wheel
- Roll/pitch earth sensors
- Horizon sensors
- Yaw rate gyro
- Roll torquer
- Sun sensor
- Bias coils

Propulsion

- Solid propellant apogee motor
- Four-tank $N_2H_4$ system
- 12 valves/thrusters

Table C.24-1. RCA SatCom Subsystem Description (Cont.)

Electrical Power

- Batteries
- Solar arrays (7702 cells)
- Array drive
- Regulation equipment
- 740 W beginning of life

Thermal Control

- Sensors
- Blankets
- Radiators

Communications

- Antenna
- Receivers
- TWTAs (24)
- 24 channels in 500 MHz band

Command, Ranging, and Telemetry

- Beacon transmitter
- Command receiver
- Omni antenna
- 248 command capability
- 128 telemetry parameter capability

## C.25 JAPANESE BROADCAST SATELLITE (GE B.S.E.)

### C.25.1 Program Summary

The Broadcast Satellite is being designed and developed by the General Electric Company Space Division in Valley Forge, Pennsylvania, for the Japanese Space Agency. Toshiba of Japan is performing subsystem design tasks and providing technical management support. The satellite is experimental, its purpose being to evaluate TV/voice transmission/reception and to establish mission control techniques.

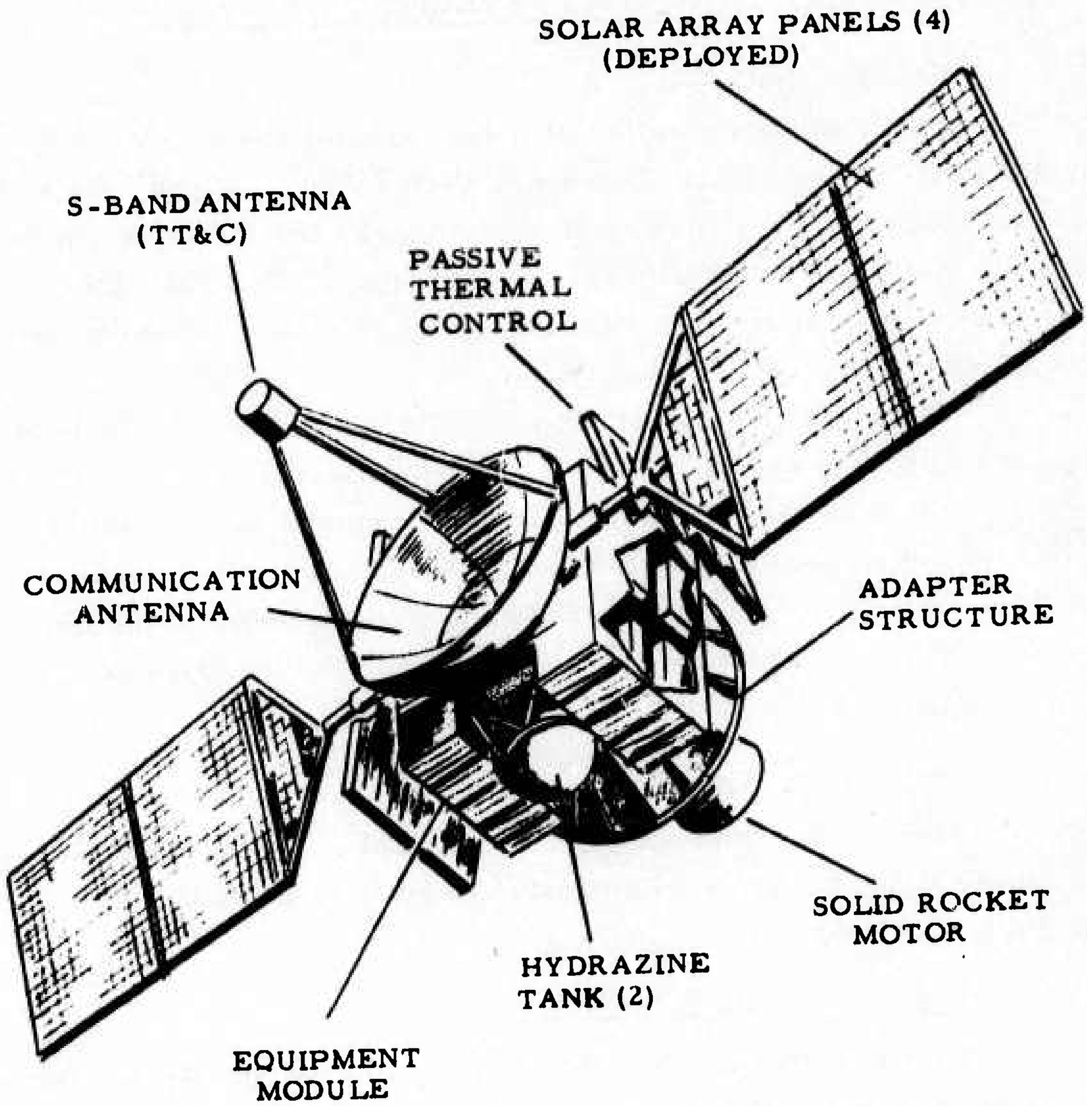
The program, initiated in early 1974, consists primarily of one protoflight (qualification and flight) satellite to be delivered in 1976. The satellite will be launched on a Delta 2914 booster and positioned at 110°E in synchronous equatorial orbit. The antenna is designed to provide primary coverage of the main Japanese islands and secondary coverage of the Okinawa and Ogasawara islands. Additional general details of the program can be found in Reference C.25-1.

### C.25.2 Satellite Description

The configuration and general features of the satellite are shown in Figure C.25-1. Additional details are given in Table C.25-1 (Refs. C.25-1, C.25-2).

### C.25.3 Key Events and Milestones

Primary milestones of the program through the planned launch are indicated in Figure C.25-2 (Refs. C.25-2, C.25-3).




CHARACTERISTICS

| | | |
|---|---|---|
| LAUNCH DATE | - | MAR 1978 |
| BOOSTER | - | DELTA 2914 |
| ORBIT | - | SYNC. EQUATORIAL |

| | | |
|---|---|---|
| WIDTH (ASCENT) | - | 5.2 FT |
| WIDTH (DEPLOYED) | - | 30 FT |
| HEIGHT | - | 10 FT |
| WT (LIFTOFF) | - | 1475 LB |
| POWER (BOL) | - | 1000 WATTS |
| DESIGN LIFE | - | 3 YEARS |

Figure C.25-1. Japanese Broadcast Satellite (GE B.S.E.)

Table C.25-1. Japanese Broadcast Satellite Technical Details

Satellite

- 30-ft width (deployed), 10-ft height
- 1475 lb
- Three-axis stabilization

Power

- Four deployed solar arrays, 1000 W beginning of life, 720 W after 3 years
- Battery for eclipse operation

Attitude Control

- Three-axis on-orbit control based on Nimbus/ERTS system
    - Three reaction wheels for control torquing
    - Earth sensor, sun sensors, and RF monopulse for attitude determination
    - 0.2-lb thrusters for unloading momentum of reaction wheels
- Spin stabilization during transfer orbit and orbit injection
    - Earth and sun sensors for attitude determination
    - Passive damping for nutation control
    - 5-lb thrusters for precession
    - Yaw rate gyro for despin

Communications

- Single-conversion transponder
- Two TV channels
- Channel A - 14.25 to 14.3 GHz receive,
  11.95 to 12.0 GHz transmit
- Channel B - 14.35 to 14.43 GHz receive,
  12.05 to 12.13 GHz transmit
- Two 100-W TWTs

Table C.25-1. Japanese Broadcast Satellite Technical Details (Cont.)

Tracking, Telemetry, and Command

- Combined Ku-band and S-band system
  - 394 TLM points
  - 246 commands
- S-band compatible with STDN (USB)
- S-band and Ku-band provide backup for each other

Antennas

- 3.4 × 5.2 ft communications antenna
  - 1.3° × 2.3° beamwidth
  - +37 dB gain (edge of main Japanese islands)
  - Designed for rapid fall-off outside coverage area (not to exceed +28 dB toward China, Korea, Soviet Union)
  - Uses Kelvar epoxy material to minimize weight and reduce thermal distortions
- S-band tracking, telemetry, and command antenna

Design Life

- 3 years

Launch Data

- Launch March 1978
- Delta 2914 launch vehicle
- Synchronous equatorial orbit

Developed by

- General Electric Company
- Japanese Space Agency

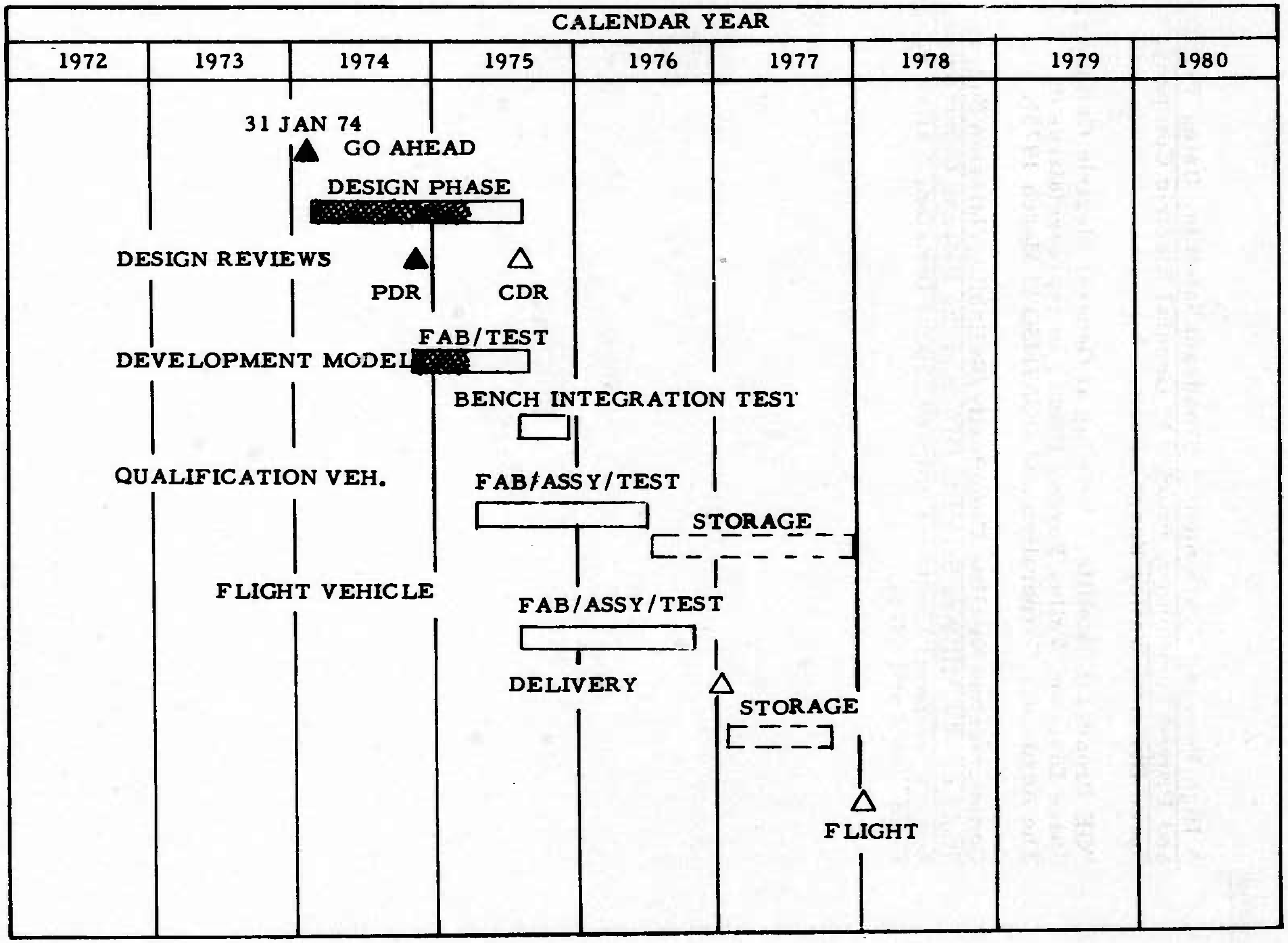



Figure C. 25-2. Key Milestones and Events - Japanese Broadcast Satellite (GE B.S.E.)

References

C.25-1. A High Powered Experimental Broadcast Satellite, Using Tried and Proved Technology, brochure, General Electric Company, Space Division, Valley Forge, Penn.

C.25-2. "GE Broadcast Satellite," briefing at General Electric Company, Space Division, Valley Forge, Penn., to representatives of The Aerospace Corporation and DCA/MSO (3 March 1975).

C.25-3. Communication Satellite Complexity/Reliability/Mission Success Study for the Military Satellite Office of the Defense Communications Agency, General Electric Company, Space Division, Valley Forge, Penn. (28 April 1975).

## C.26 SYMPHONIE COMMUNICATIONS SATELLITE

### C.26.1 Program Summary

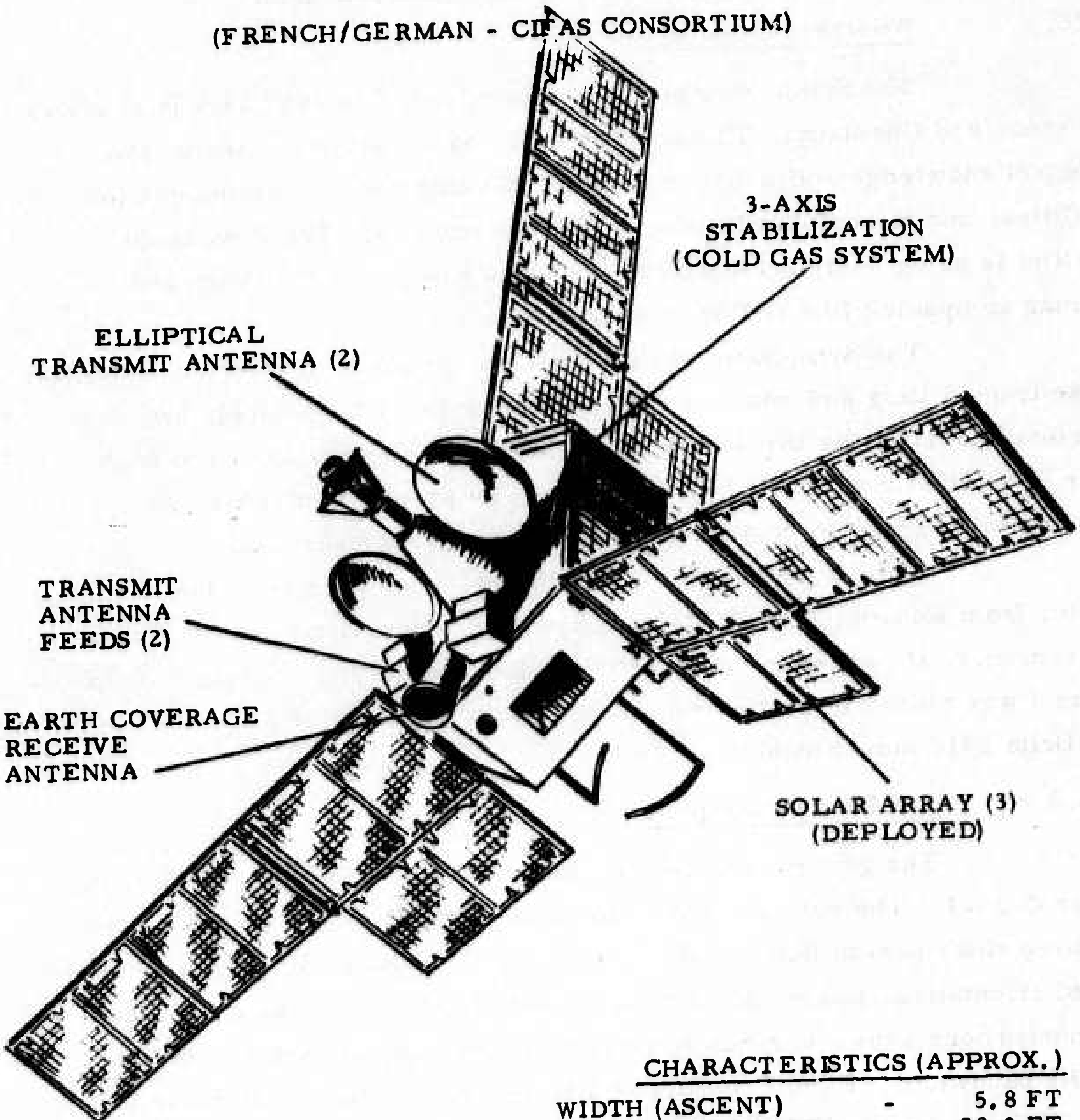
The Symphonie program was initiated in 1967 as a joint effort of France and Germany. The primary goals of the program are to gain technical knowledge and experience in the development of communication satellites, and to perform transmission experiments. The Symphonie satellite is being designed and developed by a group of six French and German companies (the CIFAS consortium).

The Symphonie system plan was to place in orbit two satellites whose transmitting and receiving frequencies were not identical, but interleaved. Thus the two satellites could be placed very closed to each other in orbit without mutual interference. To ground terminals they would almost appear to be a single satellite with four channels.

Launch plans were originally based on the use of the Europa II booster from Kourou, French Guiana; however, cancellation of the Europa II program made it necessary to launch Symphonie on a U.S. booster. Symphonie 1 was placed in a synchronous equatorial orbit on 18 December 1974 by a Delta 2914 launch vehicle.

### C.26.2 Satellite Description

The general features of the Symphonie satellite are shown in Figure C.26-1. The satellite has a three-axis-stabilized hexagonal body and three solar panels that are deployed in orbit. The solar panels maintain a fixed orientation, not having any mechanism for tracking the sun. The communications subsystem has two double-conversion channels with a 90-MHz bandwidth. Each channel has a tunnel diode preamplifier and a 13-W TWT transmitter. A single earth-coverage horn is used for reception. Two elliptical reflectors with off-axis feeds are used for transmission. Each reflector will produce an 8° × 13° beam. One TWT is connected to each transmitting antenna; a switch allows reversing these connections. It




LAUNCHES

SYMPHONIE 1 - 18 DEC. 1974

BOOSTER - DELTA 2914

ORBIT - SYNC. EQUATORIAL

CHARACTERISTICS (APPROX.)

| | | |
|---|---|---|
| WIDTH (ASCENT) | - | 5.8 FT |
| WIDTH (DEPLOYED) | - | 23.0 FT |
| HEIGHT (ASCENT) | - | 4.0 FT |
| HEIGHT (DEPLOYED) | - | 5.8 FT |
| WEIGHT (LIFTOFF) | - | 1000 LB |
| POWER (BOL) | - | 300 WATTS |
| DESIGN LIFE | - | 5 YEARS |

Figure C.26-1. Symphonie Communications Satellite is intended to station the satellites over the Atlantic Ocean, with one transmitting antenna covering most of Europe and Africa, and the other covering the Eastern U.S. and Canada and part of South America. Other details of the Symphonie satellite are given in Table C.26-1.

Table C.26-1. Symphonie Technical Details

Satellite

- Hexagonal body, 68 in. maximum diameter, 20 in. high, 23 ft diameter with solar panels deployed
- 480 lb
- Solar cells and NiCd batteries, 300 W initial, 167 W minimum after 5 years (batteries do not support the communication subsystem during eclipse)
- Three-axis stabilization, 0.5° attitude control accuracy

Configuration

- Two 90-MHz-bandwidth double-conversion repeaters

Capacity

- 600 one-way voice circuits or 1 color TV signal with 3 voice channels per repeater

Transmitter

- Satellite 2 - 3715 to 3805 and 3970 to 4060 MHz
- Satellite 1 - 3840 to 3930 and 4095 to 4185 MHz
- Single 13-W TWT per channel (no redundancy)
- ERP - 29 dBW minimum per channel over 8° × 13° field of view

Receiver

- Satellite 2 - 5940 to 6030 and 6195 to 6285 MHz
- Satellite 1 - 6065 to 6155 and 6320 to 6410 MHz
- Tunnel diode preamplifier
- ~7.5 dB noise figure
- -15 dB/°K G/T[a] minimum over 17° field of view

[a]Gain/temperature ratio

Table C.26-1. Symphonie Technical Details (Cont.)

Antenna

- Receive - earth coverage horn, 17.2° beamwidth, circular polarization
- Transmit - 2 elliptical reflectors with offset feeds (one per channel), 8° × 13° beamwidth, circular polarization

Design Life

- 5 years

Orbit

- Synchronous equatorial, 11.5°W longitude

Launch Record

- Satellite 1 - December 1974
- Satellites 2 & 3 - 1975
- Delta launch vehicle

Developed By

- French and German national space agencies
- CIFAS (French-German industrial consortium)

## C.27 APPLICATIONS TECHNOLOGY SATELLITES (ATS)

### C.27.1 General Summary

The NASA Applications Technology Satellite (ATS) program evolved from the Advanced Syncom study. Basic communication satellite development was continued with the Early Bird program, while communication experiments were put into the ATS program. ATS also included meteorological, attitude control and stationkeeping, space environment, and other experiments.

The program has encompassed two generations of spacecraft. The first five satellites in the program (designated ATS 1 to ATS 5) formed the first generation; ATS 6 (ATS F)* the second generation.

### C.27.2 ATS 1 to 5

#### C.27.2.1 Program Summary

The ATS 1 to 5 satellites, built for NASA by Hughes Aircraft Company, were launched on Atlas or Atlas/Centaur boosters, the first launch being in 1966. Of the five ATS launches, three satellites were successfully placed into orbit. ATS 2 and 4 did not achieve the desired orbit because of launch vehicle malfunctions, and little experimental data was obtained. The ATS 2 C-band repeaters operated 12 and 626 hours; the ATS 4 repeaters operated only 9 and 30 hours. ATS 4 was in orbit only 2 months; ATS 2 was in orbit over 2 years but was deactivated after 6 months.

The experiments on both ATS 1 and ATS 3 were used extensively after the satellites were in orbit. Through March 1971 the four microwave communication repeaters on these satellites had accumulated about 35,000 hours of use. Tests were run in all modes, and numerous spacecraft parameters were measured. Various tests were run to determine the values

---

*ATS F was the spacecraft designation of the ATS 6 satellite prior to launch. The ATS 1 to 5 satellites had similar letter designations prior to launch, i.e., ATS A to E.

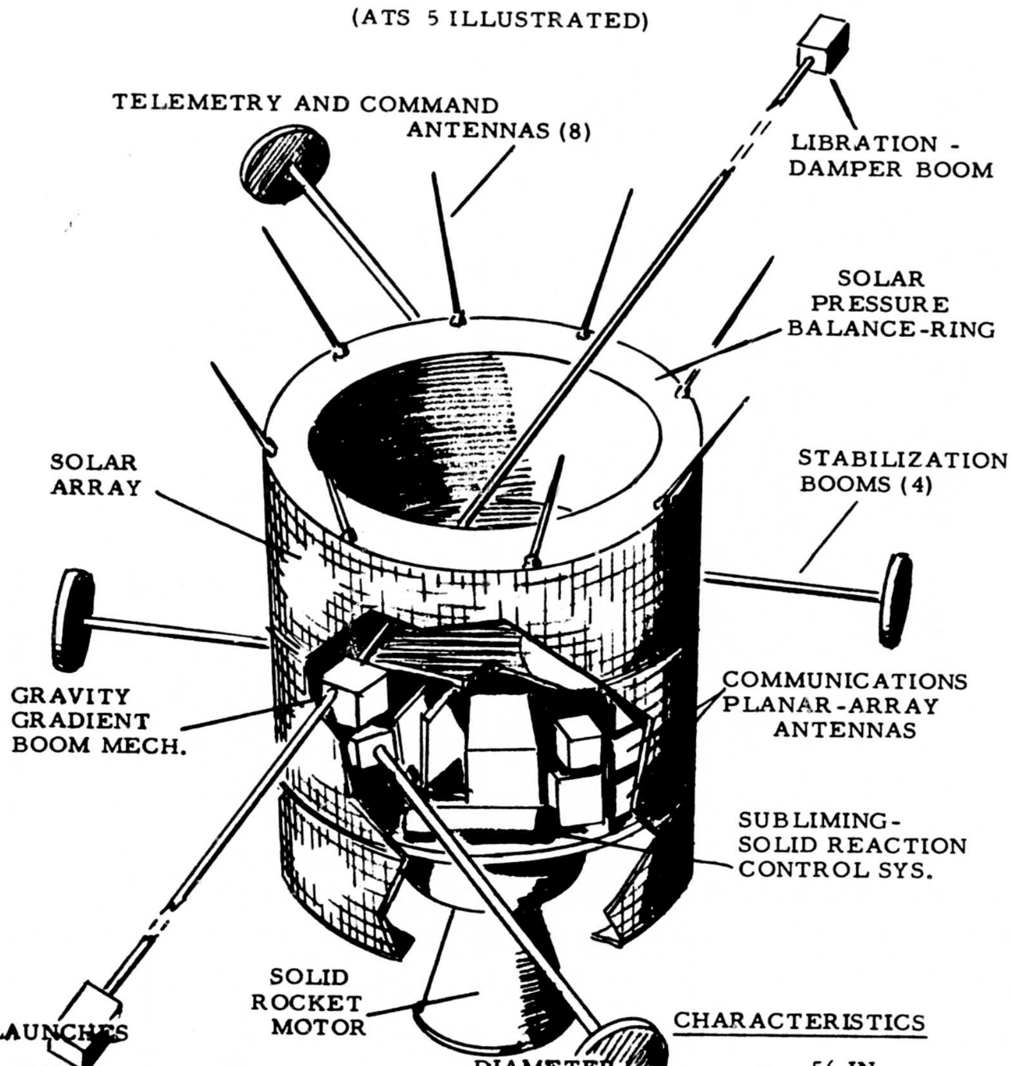
of system noise, delay, frequency response, intermodulation, etc. In general, the system performance was satisfactory according to commercial standards. The C-band communications equipment was also used a number of times for international television broadcasts of public interest.

Engineering performance measurements were also made on the VHF equipment. System performance was evaluated for ground-satellite-aircraft links using equipment installed on several commercial aircraft. The Coast Guard performed tests using several shipborne terminals. In general the results with both aircraft and ships were fair to good communications; the quality of the satellite link was usually as good or better than alternate communication links. The VHF equipment was also used for experiments in clock synchronization, navigation, and meteorological data collection and dissemination. Results were varied, often limited by available equipment or satellite design, but the experiments did provide a data base and recommendations for future work.

ATS 5 was successfully placed into synchronous orbit. The satellite was to be spinning upon orbital injection and then despun, at which time the gravity gradient stabilization would begin. However, during orbital injection the satellite developed a spin about an axis normal to the intended spin axis. In this orientation the satellite could not be despun. Because of the spinning condition, the satellite antennas point toward the earth only a small portion of each revolution. The communication experiments have been operated with limited success in a pulsed type of operation synchronized with the periods of correct antenna orientation.

### C.27.2.2 Satellite Description

The configuration and general characteristics of the ATS 1 to 5 satellites are shown in Figure C.27-1. The five satellites have some basic similarities, which are summarized in Table C.27-1. The table also indicates which communication experiments were in each satellite. Details of those experiments are given in Table C.27-2.




**LAUNCHES**

| | |
|---|---|
| ATS 1 | - 7 DEC 66 |
| ATS 2 | - 6 APR 67 |
| ATS 3 | - 5 NOV 67 |
| ATS 4 | - 10 AUG 68 |
| ATS 5 | - 12 AUG 69 |

**CHARACTERISTICS**

| | |
|---|---|
| DIAMETER | - 56 IN |
| HEIGHT | - 72 IN |
| WEIGHT (LIFTOFF) | - 1730 LB |
| POWER (BOL) | - 130 WATTS |
| DESIGN LIFE | - 3 YEARS |

BOOSTER - ATLAS & ATLAS/CENTAUR

ORBIT - SYNC. EQUATORIAL (ATS 1, 3, 5)

Figure C.27-1. Applications Technology Satellites (ATS 1 - 5)

Table C.27-1. ATS 1 to 5 Technical Details

| Satellite[a] | ATS 1 (B) | ATS 2 (A) | ATS 3 (C) | ATS 4 (D) | ATS 5 (E) |
|---|---|---|---|---|---|
| Cylinder | | | | | |
| Diameter, in. | 58 | 56 | 58 | 56 | 56 |
| Height, in. | 54 | 72 | 54 | 72 | 72 |
| Weight, lb | 775 | 702 | 775 | 670 | 670 |
| Solar cells and NiCd batteries, | | | | | |
| Initial power, W | 175 | 130 | 175 | 130 | 130 |
| Stabilization | spin | gravity gradient | spin | gravity gradient | gravity gradient |
| Design life, years | 3 | 3 | 3 | 3 | 3 |
| Actual orbit | sync. eq. | 100 × 6000 nmi | sync. eq. | 130 × 480 nmi | sync. eq. |
| (Intended orbit) | | (6000 nmi circ.) | | (sync. eq.) | |
| Launch date | 7 Dec 1966 | 6 April 1967 | 5 Nov 1967 | 10 Aug 1968 | 12 Aug 1969 |
| (Decay date) | | (2 Sept 1969) | | (17 Oct 1968) | |
| Experiments | | | | | |
| C-band communication | yes | yes | yes | yes | yes |
| VHF communications | yes | | yes | | |
| Millimeter wave propagation | | | | | yes |
| L-band communications | | | | | yes |

[a] Alphabetic designations are used before launch, numeric after

Table C.27-2. ATS Experiment Details

C-band Communications (ATS 1 to 5)

Configuration

- Two 25-MHz-bandwidth repeaters

Capacity

- 1200 one-way voice circuits or 1 color TV channel

Transmitter

- 4120 and 4179-MHz center frequencies
- Two TWTs per repeater, used singly or together
- 4-W output, except 12 W at 4179 MHz on ATS 3
- ERP
    - ATS 1 - 19.5, 22.0 dBW (1, 2 TWTs)
    - ATS 3 - 22.0, 25.0 dBW (1, 24-W TWTs); 26.5 dBW (1 12-W TWT)
    - ATS 5 - 22.5, 25.0 dBW (1, 2 TWTs)

Receiver

- 6212 and 6301-MHz center frequencies
- Tunnel diode preamplifiers
- 6.2-dB noise figure

Antenna

- ATS 1 transmit - phased array, 16 sets of 4 colinear dipoles, 14-dB gain, 17° (north-south) × 21° (equatorial plane) beamwidth
- ATS 1 receive - 6-element colinear array, 6-dB gain
- ATS 2 - horn, 10.5-dB gain

Table C.27-2. ATS Experiment Details (Cont.)

- ATS 3 - mechanically despun cylindrical reflector with linear feed on cylinder (and spin) axis, 18-dB gain, 17° beamwidth
- ATS 4 and 5 receive - planar array, 4 slots in each of 4 waveguide sections, 16.3-dB gain, 23° beamwidth
- ATS 4 and 5 transmit - similar array, 16.7-dB gain

VHF Communications (ATS 1 and 3)

Configuration

- 100-kHz-bandwidth double-conversion repeater

Transmitter

- 135.6 MHz
- ATS 1 - 5 W per element, 40 W total, 22.5-dBW ERP
- ATS 3 - 6.25 W per element, 50 W total, 25.2-dBW ERP

Receiver

- 149.2 MHz
- ATS 1 - 4.5-dB noise figure
- ATS 3 - 4.0-dB noise figure

Antenna

- Eight-element (dipoles) phased array
- ATS 1 - 9-dB gain
- ATS 3 - 10-dB gain

Table C.27-2. ATS Experiment Details (Cont.)

Millimeter Wave Propagation (ATS 5)

Transmitter

- 15.3 GHz
- Solid state
- 200-mW output

Receiver

- 31.65 GHz
- 15-dB noise figure

Antenna

- Two horns, one each for transmit and receive
- 20° beamwidth, 19-dB gain

Modulation (uplinks and downlinks)

- Phase modulation, 1.43 index to provide approximately equal power in carrier and first sidebands
- Modulation frequency - none, 100 kHz, 1 MHz, 10 MHz, or 50 MHz

L-band Communications (ATS 5)

Configuration

- 25-MHz-bandwidth repeater

Transmitter

- 1550-MHz center frequency
- Two TWTs used singly or together
- 12 W per TWT, 22.4-dBW ERP (one TWT), 25.4-dBW ERP (two TWTs)

Receiver

- 1651-MHz center frequency
- 8-dB noise figure

Antenna

- 17.2-dB gain

C.27.3 ATS 6 (ATS F)

C.27.3.1 Program Summary

The ATS 6 satellite (also known by its prelaunch designation ATS F) is built by Fairchild Industries. This second generation satellite in the NASA Applications Technology Satellite program contains 12 technology experiments, eight of which are communications and propagation studies that extend over a frequency range from 860 MHz to 30 GHz.

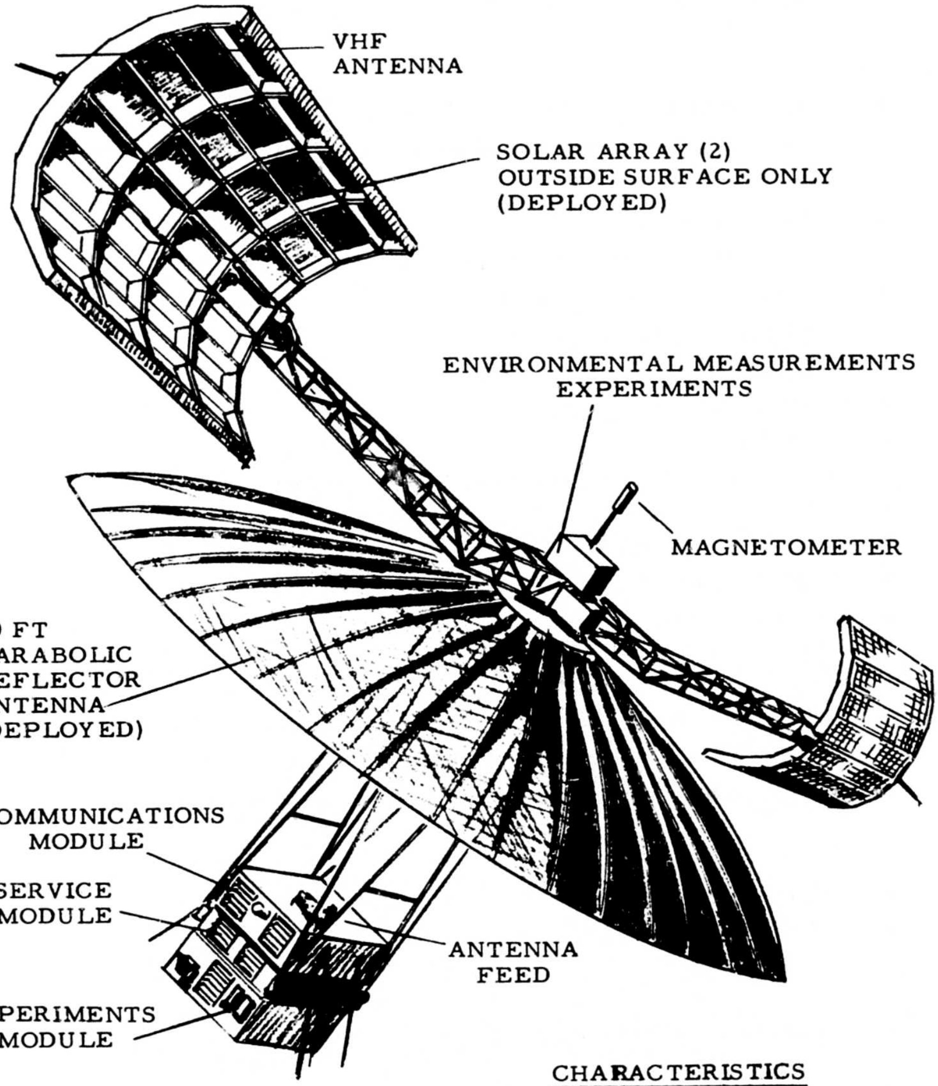
The ATS 6 was launched on 30 May 1974 by a Titan IIIC booster into a synchronous equatorial orbit and successfully initiated on-orbit operations.

The original program for ATS F included a second satellite of the same design, designated ATS G. This satellite was cancelled due to NASA budget restrictions, but the satellite components are available for possible assembly and launch.

C.27.3.2 Satellite Description

The configuration and general features of ATS 6 are shown in Figure C.27-2. The satellite consists of a 30-foot-diameter parabolic antenna, an earth-viewing module located at the focus of the parabola, two solar arrays, and the interconnecting structures. The antenna and the solar arrays are deployed after the satellite is in orbit. All the communications experiments are located in a section of the earth-viewing module. Feed horns for the large parabola are mounted on top of the module; other antennas are on the bottom. Satellite characteristics are summarized in Table C.27-3. Details of the communications experiments are given in Table C.27-4.

The communications equipment on ATS 6 is composed of four receivers (C-, S-, L-band and 13/18 GHz), three IF amplifiers, and five transmitters (C-, S-, L-band, 860 MHz and 20/30 GHz). The 13/18 GHz uplink is downconverted to C-band, amplified, and routed to the C-band transmitter. The other uplinks are amplified and filtered before downconversion to the 150 MHz intermediate frequency. Any receiver (except 13/18 GHz) may be connected to any one of the three identical IF amplifiers, which can provide either 12 or 40 MHz bandwidths. The IF outputs may be connected to any of




LAUNCH DATE - 30 MAY 1974

BOOSTER - T-IIIC

ORBIT - SYNC. EQUATORIAL

CHARACTERISTICS

| | | |
|---|---|---|
| DIAMETER (ASCENT) | - | 54 IN |
| WIDTH (DEPLOYED) | - | 51.6 FT |
| HEIGHT (OVERALL) | - | 27.5 FT |
| WEIGHT (LIFTOFF) | - | 2950 LB |
| POWER (BOL) | - | 645 WATTS |
| DESIGN LIFE | - | 5 YEARS |

Figure C.27-2. Applications Technology Satellite (ATS 6/ATS F)

## Table C.27-3. ATS 6 (ATS F) Technical Details

Structure

- 30-ft-diameter parabolic reflector, 6-1/2-ft-diameter hub section with copper-coated dacron mesh supported by 48 aluminum ribs
- Earth-viewing module at antenna focus with experiment sections and support subsystems, 54 × 54 × 65 in.
- Two solar arrays (deployed in space), each half of a cylinder, 54 in. radius, 94 in. long
- Maximum height 27 ft 6 in.
- Maximum span 51 ft 8 in.
- 2950 lb

Power

- Solar cells and NiCd batteries
- 645 W maximum, initial
- 415 W minimum, 5 years

Stabilization

- Three-axis-stabilized, 0.1° pointing accuracy
- Pointing to any location on earth
- Tracking of low-altitude satellite over ±11° from local vertical

Design Life

- 2 years (required)
- 5 years (goal)

Orbit

- Synchronous equatorial, 94° W longitude

Launch Date

- 30 May 1974
- Titan IIIC launch vehicle

Developed by

- NASA
- Fairchild Industries

Table C.27-4. ATS 6 (ATS F) Experiment Details

Position Location and Aircraft Communication Experiment (PLACE)

Configuration

- Two-way link through ATS 6 between a ground terminal and aircraft both voice and ranging functions

Transmitter (ATS 6 to aircraft link)

- 1550 MHz
- 40-W output, 40.3- or 51.0-dBW ERP

Receiver (aircraft to ATS 6 link)

- 1650 MHz
- -4.4 or +5.5 dB/°K G/T[a]

Antenna

- 30-ft parabola, 28- to 29-dB gain with 0.8° × 7.5° fan beam, 38.5-dB gain with 1.5° pencil beam, circular polarization

Transmitter (ATS 6 to ground link)

- One of 3750, 3950, or 4150 MHz
- 12-W output, 28-dBW ERP on axis

Receiver (ground to ATS 6 link)

- One of 5950, 6150, or 6350 MHz
- -17 dB/°K peak G/T

Antenna

- Horn, 16.3- to 16.5-dB gain, 13° × 20° beamwidth, linear polarization

[a]Gain/temperature ratio

Table C.27-4. ATS 6 (ATS F) Experiment Details (Cont.)

Satellite Instructional Television Experiment (SITE)

Configuration

- 40-MHz-bandwidth double-conversion repeater
- Frequency modulation

Transmitter

- 860 MHz (3750 MHz to be used occasionally to monitor signals)
- 80-W output, 51.0-dBW ERP peak

Receiver

- 5950 MHz
- -17 dB/°K peak G/T

Antenna

- Transmit - 30-ft parabola, 33-dB peak gain, 2.8° beamwidth, circular polarization at 860 MHz
- Receive - Horn, 16.3-dB peak gain, 13° × 20° field of view, linear polarization (30-ft parabola might be used for receiving instead of horn, 48.4-dB peak gain, 0.4° bandwidth, +13.7 dB/°K G/T)

Table C.27-4. ATS 6 (ATS F) Experiment Details (Cont.)

## Television Relay Using Small Terminals (TRUST)

(Satellite parameters are the same as for SITE)

## Educational Television (ETV)

### Configuration

- Forward link - two 30-to 40-MHz bandwidth repeaters for two FM-TV carriers with sound subcarriers plus separate telephone carriers
- Return link - for telephone link, parameters not yet defined

### Transmitter

- 2569 and 2670 MHz (also C-band for monitoring)
- 15-W output, 53.0-dBW peak ERP

### Receiver

- Same as for SITE

### Antenna

- Transmit - 30-ft parabola, 41.5-dB peak gain, 1° beamwidth, circular polarization
- Receive - Same as for SITE

Table C.27-4. ATS 6 (ATS F) Experiment Details (Cont.)

<u>Tracking and Data Relay Experiment</u>

<u>Configuration</u>

- Two 12- or 40-MHz-bandwidth channels
- Two-way link through ATS 6 between ground and a low-altitude satellite

<u>Transmitter (ATS 6 to satellite link)</u>

- 2063 MHz
- 20-W output, 48.0-dBW ERP minimum

<u>Receiver (satellite to ATS 6 link)</u>

- 2253 MHz
- 7.0 dB/°K minimum G/T

<u>Antenna (satellite to satellite links)</u>

- 30-ft parabola, 36.4-dB gain minimum, 13.2° overall field of view using switched feeds, circular polarization

<u>Transmitter (ATS 6 to ground link)</u>

- 3753 MHz primary (alternates 3953 or 4153 MHz)
- 12-W output, 28.0-dBW ERP peak

<u>Receiver (ground to ATS 6 link)</u>

- 5938 MHz primary (alternates 6138 or 6338 MHz)
- -17 dB/°K peak G/T

<u>Antenna (satellite to ground links)</u>

- Horn, 16.5-dB transmit gain (peak), 16.3-dB receive, 13° × 20° field of view, linear polarization

Table C.27-4. ATS 6 (ATS F) Experiment Details (Cont.)

C-Band RFI

Receiver

- 5925 to 6425 MHz
- +17.0 dB/°K G/T (30-ft parabola) or -17.0 dB/°K G/T (horn) peak, minimum detectable ground source is 10-dBW ERP

Antenna

- 30-ft parabola, 48.4-dB gain peak, 0.4° beamwidth, circular or linear polarization
- Horn, 16.3-dB gain peak, 13° × 20° beamwidth, linear polarization

Millimeter Wave Experiment (NASA)

Configuration

- Propagation modes - CW or multitone downlinks
- Communications mode - 40-MHz-bandwidth repeater

Transmitter (propagation modes)

- 20.0 and 30.0 GHz
- CW - 2-W output, 30-dBW peak ERP
- Multitone (9 tones) - 0.06-W output/tone, 17.8-dBW peak ERP/tone

Transmitter (communications mode)

- 20.15 and 30.15 GHz and one of 3750, 3950, or 4150 MHz
- 20.15 GHz - 2-W output, 40-dBW peak ERP
- 30.15 GHz - 2-W output, 42-dBW peak ERP
- C-band - 12-W output, 28-dBW peak ERP

Table C.27-4. ATS 6 (ATS F) Experiment Details (Cont.)

Receiver (communications mode only)

- One of 5950, 6150, or 6350 MHz
- 13.7 dB/°K G/T (30-ft parabola), -17 dB/°K G/T (horn)

Antenna

- Propagation mode - horn, 27-dB peak gain, 5° × 7° beamwidth, linear polarization
- Communication mode
  - 20.15 GHz - 1.5-ft parabola, 37-dB gain, 2.4° beamwidth
  - 30.15 GHz - 1.5-ft parabola, 39-dB gain, 1.6° beamwidth
  - C-band transmit - horn, 16.5-dB gain, 13° × 20° beamwidth
  - C-band receive - horn, 16.3-dB gain, 13° × 20° beamwidth or 30-ft parabola, 48.4-dB gain, 0.4° beamwidth

Millimeter Wave Experiment (Comsat Corp.)

Configuration

- 39 unmodulated uplink carriers received and retransmitted to a control ground terminal in a 30-MHz bandwidth

Transmitter

- 4150 MHz
- 0.2- to 1.3-mW output per carrier
- -13 to -21 dBW ERP per carrier

Receiver

- 15 carriers near 13.19 GHz and 24 carriers near 17.79 GHz
- 10-dB noise figure

Antenna

- Transmit - horn, 17-dB gain
- Receive - 1-ft parabola, 25.3/28.0-dB peak gain (13/18 GHz), 4° × 8° beamwidth, linear polarization

the transmitters. The transmitters include upconverters, driver amplifiers, and power amplifiers; most of the transmitter elements are redundant. The C-band transmitter uses TWTs while the lower frequency transmitters are all transistorized. The primary communication antenna is the 30-foot parabola; in addition, the satellite has a C-band horn, and a small parabola and two horns for the millimeter wave experiments. The feed structure for the large reflector includes 36 elements to provide efficient performance for the various frequencies and beam patterns used in the communications experiments.

## C.27.4 Key Events and Milestones

Significant milestones for all satellites in the ATS program are shown in Figure C.27-3. Major launch and on-orbit malfunctions are also indicated.

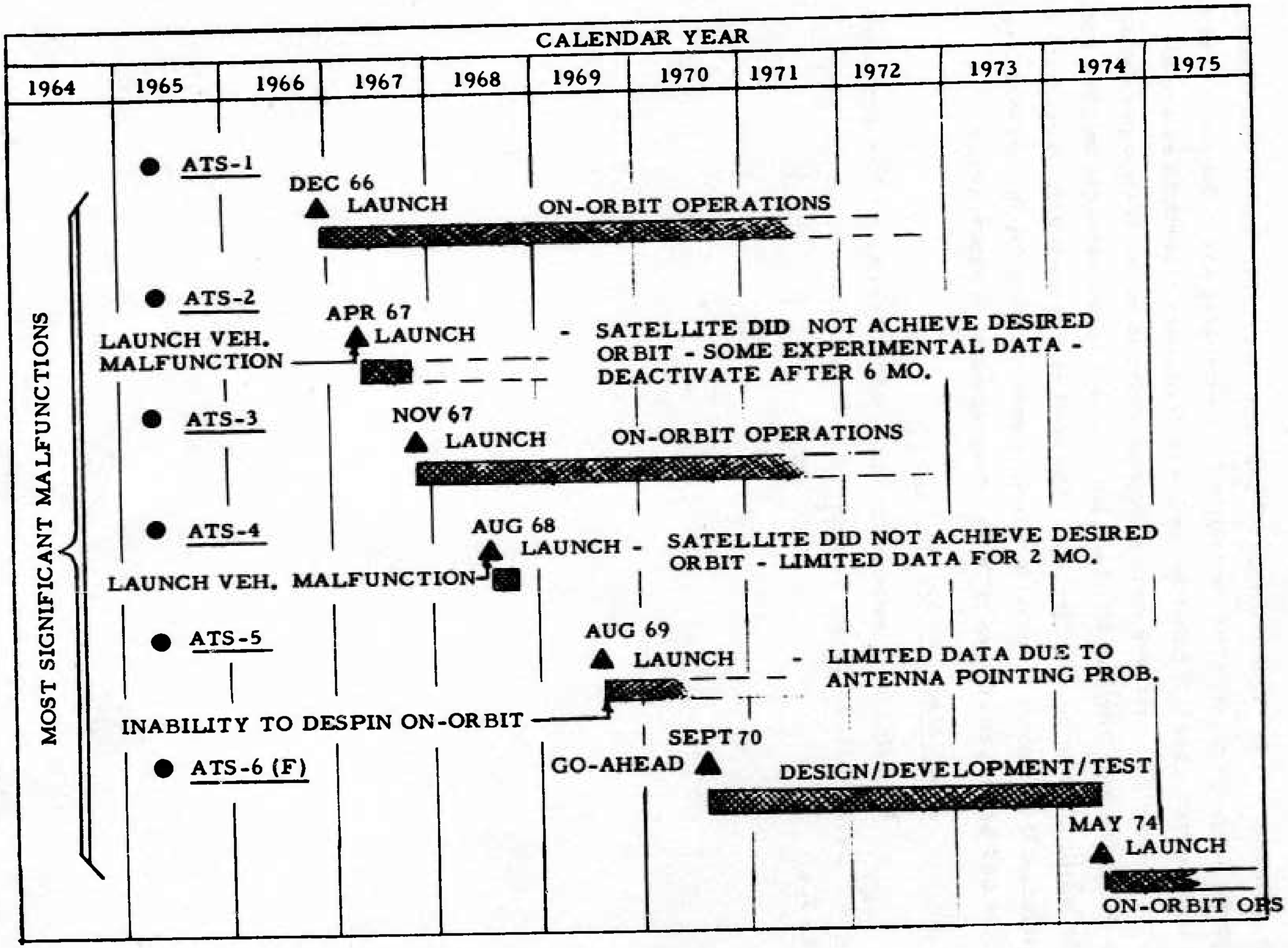



Figure C. 27-3. Key Milestones and Events - ATS Program

## C.28 NIMBUS METEOROLOGICAL SATELLITE

### C.28.1 Program Summary

When the Nimbus 1 (Nimbus A) meteorological satellite was launched into orbit in 1964, it carried three experiment packages and weighed 820 lb. Each succeeding spacecraft has grown in sophistication, complexity, weight, application, and performance. Nimbus 4 (Nimbus D), launched in 1970, carried nine experiment packages and weighted 1366 lb. Nimbus 5 (Nimbus E), launched in 1972, carried six experimental payloads and weighed 1580 lb.

The Nimbus satellites are placed in a near-earth, sun-synchronous polar orbit to provide continuous global coverage twice in every 24-hour period, once in sunlight and the other in darkness. The growth of satellite payloads indicates the increasing complexities and capabilities of the basic satellite support functions.

General objectives of the Nimbus program are:

a. Development and flight application of advanced passive radiometric and spectrometric sensors for the daily global surveillance of the earth's atmosphere to provide a data base for long-range weather forecasting;

b. Development and evaluation of new active and passive sensors for sounding the earth's atmosphere and mapping surface characteristics;

c. Development of advanced space technology and ground techniques for meteorological and other earth-observational spacecraft;

d. Development of new techniques and knowledge useful for the exploration of other planetary atmospheres;

e. Participation in global observation programs (world weather watch) by expanding daily global weather observation capability;

f. Provision of a supplemental source of operational meteorological data.

The Nimbus program procuring agency is NASA's Goddard Space Flight Center (GSFC). The major contractor in the program is the General Electric Company Space Division, Valley Forge, Pennsylvania.

The first Nimbus contract called for three flight spacecraft. Nimbus A was launched on 28 August 1964 (Nimbus 1). Nimbus C was launched on 15 April 1966 (Nimbus 2). Nimbus B was launched 18 May 1968, but booster failure resulted in loss of the satellite.

The second contract called for basic spacecraft design studies during the period January - July 1967. This was followed by a contract calling for integration and test of Nimbus D, launched 8 April 1970 (Nimbus 4). Meanwhile, a fourth contract called for assembly and test of one flight spacecraft, Nimbus B2 (Nimbus 3), which was launched 14 April 1969.

Basic spacecraft design modifications were made before the Nimbus E and Nimbus F contract. Nimbus E was launched on 11 December 1972 (Nimbus 5). Nimbus F is scheduled for launch in mid-1975.

## C.28.2 Satellite Description

Nimbus is a butterfly-shaped satellite, 10 feet tall and 5 feet wide. The satellite, which is of a basically similar design to the ERTS/LANDSAT satellite (see Section C.31), consists of four major structural elements:

a. Attitude Control Subsystem (ACS) - Located at top of satellite.

b. Solar Array Paddles - Attached to shaft projecting from ACS solar array drive modules.

c. Truss Structure - Provides tripod connection between ACS and sensory ring assembly.

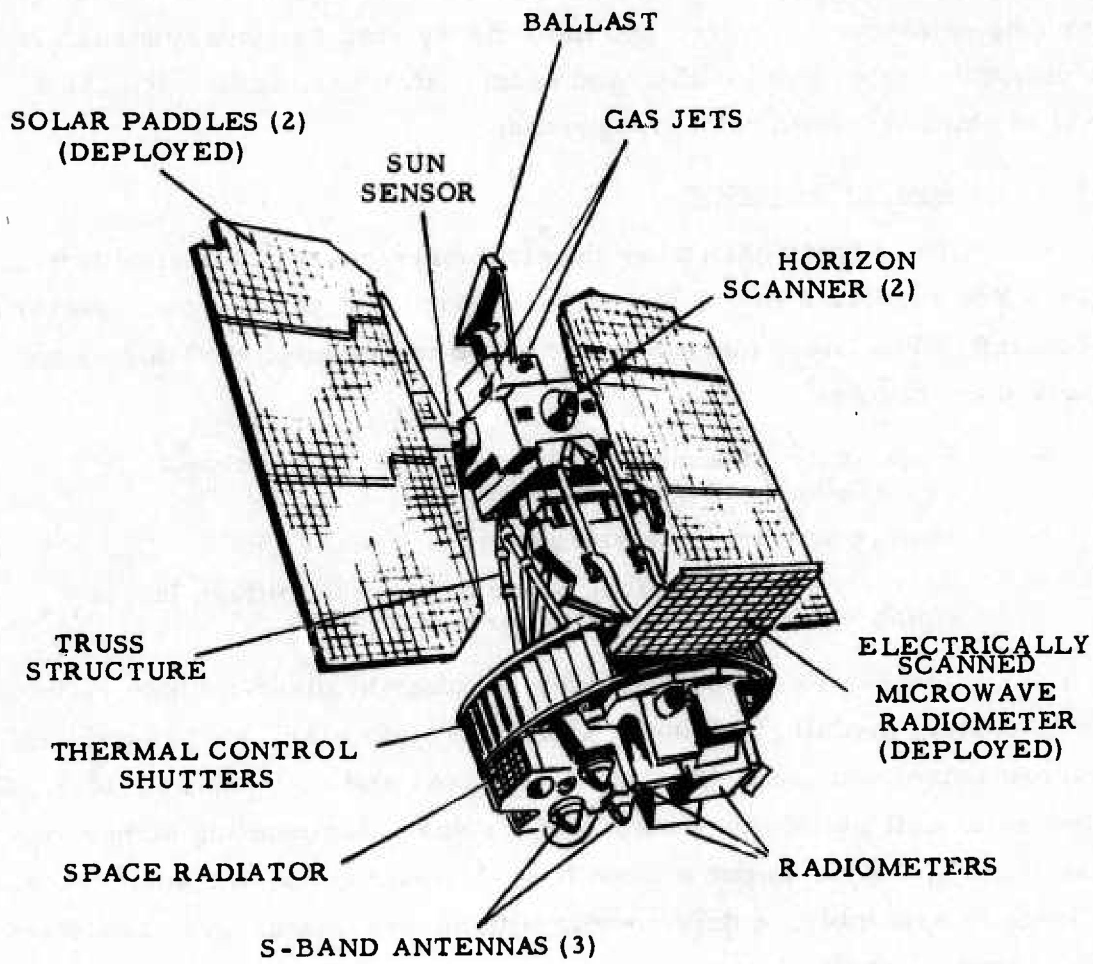
d. Sensory Ring Assembly - Houses additional service subsystems and payload experiments.

The configuration and general features of the satellite are shown in Figure Figure C.28-1. Details of specific Nimbus and ERTS subsystems are given below.

### C.28.2.1 Attitude Control Subsystem

ACS consists of four attitude-control loops and associated switching logic, telemetry, and test outputs; necessary electrical power conversion circuits; and pneumatic supply storage tank and associated manifolding.

(NIMBUS 5 SHOWN)




LAUNCH DATES

| | | |
|---|---|---|
| NIMBUS 1 | - | 28 AUG 1964 |
| NIMBUS 2 | - | 15 MAY 1966 |
| NIMBUS B | - | 18 MAY 1968 |
| NIMBUS B2 | - | 14 APR 1969 |
| NIMBUS 4 | - | 8 APR 1970 |
| NIMBUS 5 | - | 10 DEC 1972 |

BOOSTER - DELTA 900 (NIMBUS 5)

ORBIT - 600 NM 97$^{o}$ INCLINATION

CHARACTERISTICS

| | | | |
|---|---|---|---|
| WIDTH (ASCENT) | - | 60 | IN |
| WIDTH (DEPLOYED) | - | 156 | IN |
| HT (OVERALL) | - | 120 | IN |
| WEIGHT (LIFTOFF) | - | 1688 | LB |
| POWER (BOL) | - | 500 | WATTS |
| DESIGN LIFE | - | 1 | YEAR |

Figure C.28-1. Nimbus Meteorological Satellite

### C.28.2.2 Structure

The Structure Subsystem consists of the truss structure, the sensory ring primary structure, and the sensory ring secondary structure, the paddle cable cutter and squibs, and paddle latch hardware. Structural material is aluminum with some magnesium.

### C.28.2.3 Electrical Power

The Electrical Power Supply Subsystem was designed to provide -24.5 Vdc regulated to ± 0.5 V to the service and payload subsystems of the spacecraft. The basic functions performed to accomplish this may be summarized as follows:

a. Acquisition of incident solar radiation and photovoltaic conversion to electrical power.

b. Energy storage by electrochemical means.

c. Regulation of electrical power to provide voltage levels suitable for distribution to spacecraft loads.

The Power Subsystem consists of eight identical battery modules, one power-control module, two solar arrays, the auxiliary load controller, and power-management loads (five auxiliary loads and eight shunt loads). Each of the two solar cell platforms consists of a solar cell mounting structure with its array of phosphorous-doped silicon N-on-P solar cells, a transition section, a latching assembly, a drive motor with an associated gear reduction unit, and a control shaft clamp.

### C.28.2.4 Thermal Control

The Thermal Control Subsystem maintains the average sensory ring structure temperature within +20 ± 10°C. Passive control, in the form of coatings and/or insulation, is used on the large areas of the top and bottom surfaces of the sensory ring. Active control, in the form of movable insulation and coatings, is used on the peripheral areas of the sensory ring.

### C.28.2.5 Tracking, Telemetry, and Command

The tracking, telemetry, and command (TT&C) functions on the Nimbus and ERTS spacecraft are handled by separate and distinct service systems. On Nimbus flight spacecraft, these functions are handled by (a) a Real-Time Transmission Subsystem (not on Nimbus E and F), (b) Versatile Information Processor Subsystem (VIP), (c) High Data Rate Storage Subsystem (HDRSS)/S-Band Subsystem, and (d) Command Subsystem. On ERTS, these functions are handled by (a) Wideband Storage and Transmission Subsystem and (b) Narrowband Telemetry and Command Subsystem. Although a good portion of these subsystems handle mission data, they are integrally associated with the spacecraft platform as mission support hardware.

## C.28.3 Nimbus Experiments

### C.28.3.1 Nimbus 1

Nimbus 1 carried three experiements that successfully demonstrated the design concept, with all sensors performing exceptionally well:

a. APT (automatic picture transmission) - Transmitted photographic data of synoptic meteorological conditions in areas 1200 nmi square to over 300 ground stations in more than 43 countries.

b. AVCS (advanced vidicon camera subsystem) - Recorded three pictures simultaneously with overlapping field of view with an 800-line resolution. Detail of earth coverage was down to one-half mile.

c. HRIR (high-resolution infrared radiometer) - Provided nighttime IR coverage of the earth and cloud cover. HRIR augmented the daytime coverage provided by the APT and AVCS subsystems.

### C.28.3.2 Nimbus 2

Nimbus 2 carried four experiments:

a. AVCS, APT, HRIR - Repeated from Nimbus 1.

b. MRIR (medium-resolution infrared radiometer) - Measured heat balance over entire 200 million square mile area of the earth.

C.28.3.3 Nimbus 3

Nimbus 3 carried nine experiments:

a. HRIR, MRIR - Repeated from Nimbus 2.

b. IDCS (image dissector camera subsystem) - Provided daytime cloud cover pictures in real time to more than 300 APT stations.

c. IRIS (infrared interferometer spectrometer) - Provided information on the vertical structure of the atmosphere and emissive properties of the earth's surface.

d. IRLS (interrogation, recording, and location subsystem) - Located, interrogated, and received data from remote data collection stations for later relay to a control ground station.

e. MUSE (monitor of ultraviolet solar energy) - Measured ultraviolet radiation flux from the sun which could affect the earth's weather.

f. RMP (rate measuring package) - A technological experiment designed to assess the performance of a gas bearing gyro in a space environment.

g. SIRS (satellite infrared spectrometer) - Measured vertical temperature distribution of the atmosphere worldwide and was used operationally by the National Weather Service for short- and long-term weather forecasting.

h. SNAP-19 (system for nuclear auxiliary power) - Flown to assess capability of radioisotope power for space.

C.28.3.4 Nimbus 4

Nimbus 4, launched a year after Nimbus 3, carried nine experiments, primarily aimed at providing reliable, long-range computerized weather forecasting:

a. BUV (backscatter ultraviolet spectrometer) - Monitors the distribution of ultraviolet radiation backscattered from the earth's atmosphere.

b. FWS (filter wedge spectrometer) - Measures the distribution of water vapor in the earth's atmosphere.

c. IDCS, IRIS, IRLS, MUSE - Improved versions of the experiments flown on Nimbus 3.

d. SCR (selective chopper radiometer) - Determines temperature of six successive layers in the atmosphere from earth-cloud top level to a height of 40 km.

e. SIRS - Repeated from Nimbus 3. Water-vapor channels and a mirror-scanning function were added to enhance experiment capability. As in the case of Nimbus 3, the sensor supplied data daily to the National Weather Service.

f. THIR (temperature, humidity infrared radiometer) - Measures infrared radiation from earth during day and night portions of the orbit and provides pictures of cloud, three-dimensional mapping of cloud cover, temperature mapping of clouds, land and ocean surfaces, cirrus cloud content and contamination, and relative humidity.

### C.28.3.5 Nimbus 5

Nimbus 5, launched late in 1972, carried six highly advanced meteorological and geophysical experiment packages:

a. ESMR (electrically scanning microwave radiometer) - Provides the first microwave device for globally mapping thermal radiation from the earth's surface and atmosphere; detects precipitation and ice/sea differences; differentiates between water content of various vegetation or soils; and is able to "see" through clouds, which IR devices cannot do.

b. ITPR (infrared temperature profile radiometer) - Tests the feasibility and operational applications of the cloud-interference-elimination sounding techniques using simultaneous medium-resolution (20 miles) measurements.

c. NEMS (Nimbus E microwave spectrometer) - Demonstrates the capabilities of microwave sensors for measuring tropospheric temperature profiles, water vapor abundance, and water content of clouds even in the presence of many cloud types which block infrared sensors.

d. SCR (selective chopper radiometer) - Observes global temperature structure of the atmosphere at altitudes up to 50 km over an extended period of time.

e. SCMR (surface composition mapping radiometer) - Measures differences in the thermal emission characteristics of the earth's surface. By measuring these differences in thermal emission of igneous rock, for example, it is possible to identify their types such as dunite, basalt, syenite, or granite.

f. THIR - Repeated from Nimbus 4. This experiment is primarily a service system to provide cloud and water-vapor data to experimenters.

## C.28.4 Key Events and Milestones

Significant milestones in the Nimbus program are shown in Figure C.28-2. Major launch and on-orbit failures are also indicated.

## C.28.5 On-Orbit Malfunctions

Examples of on-orbit malfunctions and anomalies that occurred in the Nimbus program are given in Tables C.28-1 to C.28-3.

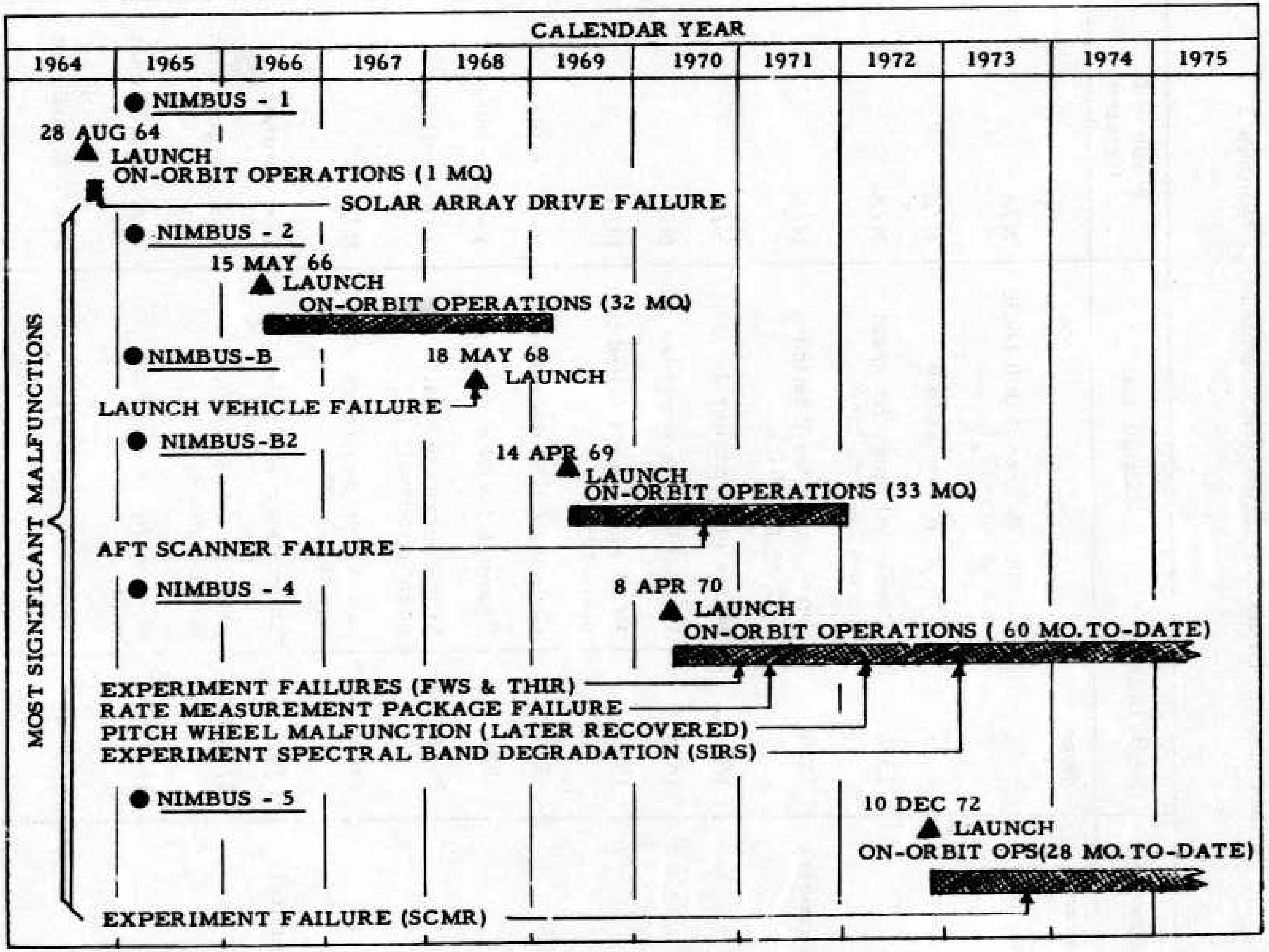



Figure C.28-2. Key Milestones and Events - Nimbus Program

Table C.28-1. On-Orbit Malfunction Summary - Nimbus 3

| Subsystem | Orbit/Date | Malfunction | Nimbus D Related |
|---|---|---|---|
| Power | None | | |
| Command | 52 | Clock "A" reset to 0 could not be reset | N/A |
| | 92 | Clock "A" reset to 0 | N/A |
| | 1332 | Clock "A" and "B" upset during IRIS failure | N/A |
| Telemetry | 10 | PCM recorder 2 failed during playback | N/A |
| | 1203 | Noise on recorder 1 | N/A |
| | 1486 | Unusable data recorder 1 | N/A |
| | 1886 | PCM recorder 1 failed to playback | N/A |
| SIRS | 519 | Channel 2 S/N decreased | Yes - minor |
| | 825 | Channel 3 S/N decreased | Yes - minor |
| | 05/13/69 | Slight increase in all channel outputs | Yes - minor |
| | 1900 | -24.5 Vdc reference voltage fluctuates | N/A |
| Thermal | 109 | Thermistor #14 failed | Yes - minor |
| IRIS | 4/16/69 | IRIS optic bolometer temps. too high | Yes - significant |
| | 4/30/69 | Neon reference signal noisy | Yes - significant |
| | 4/30/69 | Neon reference TLM continued to degrade | Yes - significant |

Table C. 28-1. On-Orbit Malfunction Summary - Nimbus 3 (Cont.)

| Subsystem | Orbit/Date | Malfunction | Nimbus D Related |
|---|---|---|---|
| IRIS (Cont.) | 1332 | IRIS failed | Yes - significant |
| MUSE | 14 125 | 1600A and 1216A sensors degraded, 2100A sensors saturated | Yes - significant |
| HDRSS | 1100 | HDRSS A flutter increase | Yes - significant |
| | 2650 | Decrease in FW/TC and IRIS HDRSS B | Yes - significant |
| | 3400 | Decrease in FW/TC and IRIS HDRSS A | Yes - significant |
| S-bands | 441 | S-band B radiating intermittently | Yes - significant |
| | 337 | S-band B power output dropped 2 PCM counts | Yes |
| Controls | 317 | SAD amplifier erratic | N/A |
| | 368 | SAD amplifier erratic | N/A |
| | 791 | Roll/yaw perturbations, sun in scanner | Yes - significant |
| | 1208 | Perturbations all axes, sun in scanner | Yes - significant |
| | 1000 | Telemetry for ACS thermal shutter #1 erratic | N/A |
| | 1345 | Yaw mode switched inadvertently back to gyro | N/A |
| | 1477 | Erratic SAD operation | N/A |
| | 1934 | Erratic SAD amplifier | N/A |

Table C.28-1. On-Orbit Malfunction Summary - Nimbus 3 (Cont.)

| Subsystem | Orbit/Date | Malfunction | Nimbus D Related |
|---|---|---|---|
| Controls (Cont.) | 2200 | Noise level increase on gyro, 200 mV → 400 mV | Yes - significant |
| | 2300 | SAD position pot became noisy | N/A |
| | 3573 | Large position perturbations and gas consumption | N/A |
| HRIR | | Sunlight entering HRIR, causing interference | Yes - significant |
| | 20 | Detector cell temp. high | Yes - significant |
| | 1138 | Eighth sync pulse amplitude problem | N/A |

## Table C.28-2. On-Orbit Anomaly Summary - Nimbus 4

| No. | Anomaly | Cause | Remarks | Nimbus E Related | Corrective Action |
|---|---|---|---|---|---|
| 1 | MUSE - Pitch aspect eye does not change at the terminator | Presently unknown | Occurred at separation | No | None |
| 2 | FWS - Ice on detector causes loss of data from long wavelength channel | Because it is the coldest spot on the spacecraft, any $H_2O$ vapor that hits it sticks as long as the local pressure is high enough | No data received up through failure in orbit 815 | Yes | None |
| 3 | IRIS - Heater operation improper | Caused by interference from the RTTS transmitter | Heater operation became proper in orbit 17 | No | None |
| 4 | IRLS - Following activation there were three unscheduled interrogations | Telemetry indicates spacecraft went into command memory load immediately after turn-on; transmissions were executions of spurious commands | This effect expected during turn-on | No | None |
| 5 | SIRS - Channel six data drops least significant 2-3 bits | Intermittently open-circuited solder joints of part leads to printed circuit boards of channel 6 synchronous demodulator | | Yes | None |
| 6 | SIRS - Apparent obstruction in field of view when CCW at 32.4°, 34.2°, and 37.8° | Caused by small protrusion of beam baffle into the optical path causing the instrument to view gold baffle, itself, and the earth's atmosphere | | Yes | None |
| 7 | BUV - Improper diffuser deployment | Believed to be the same as in test; i.e., diffuser actually deploys fully, but bounces off stops past telemetry sensor | Observed during tests | No | None |
| 8 | HDRSS - Dropout of data consistent | Selective use of bit synchronizers improved data | | Yes | None |
| 9 | SIRS - Channel 9 data failure | (Same as item 5, but for channel 9 sync demod) | | Yes | None |
| 10 | SIRS - Channel 1 data failure | (Same as item 5, but for channel 1 sync demod) | | Yes | None |
| 11 | SIRS - Channel 12 data failure | (Same as No. 5, but for channel 12 sync demod) | | Yes | None |
| 12 | IRIS - Neon ref. amp. decreased from 2.8 at launch to 2.5 by orbit 400 | Uncertain | Value at 4900 is ≈ 1.8 TMW | No | None |
| 13 | IRIS - IMCC scan decreased from 5.2° at launch to 4.4° by orbit 400 | See item 24 | | No | None |

## Table C.28-2. On-Orbit Anomaly Summary - Nimbus 4 (Cont.)

| No. | Anomaly | Cause | Remarks | Nimbus E Related | Corrective Action |
|---|---|---|---|---|---|
| 14 | Cell 6 of the primary comstor failed during orbit 525 | Unknown | Initially unable to verify beyond cell 5: modification of the PDP program now allows proper verification of all cells | Yes | |
| 15 | Cell 6 of the redundant comstor failed during orbit 611 | Unknown | Same as above | Yes | Same as above |
| 16 | HDRSS A flutter causes interference on THIR and IDCS data, also affects VIP; flutter of a lesser degree apparent on HDRSS B | Believed to be due to vibration of unsupported tape lengths excited by planetary gear noise in record | Data improved by Z axis correction circuitry | Yes | None |
| 17 | Filter wedge chopper motor failed during orbit 815 | Believed due to debris in bearings | Motor current decreased when comp. load on | No | None |
| 18 | SIRS channel 14 gain changes up to counts | Possibly same as item 5, but for channel 12 sync demodulator | | Yes | None |
| 19 | SCR cal mirror temperature failed during orbit 905 | Unknown | No detrimental effects on subsystem operation, and temperature can be inferred from cal mirror 2 | Yes | None |
| 20 | Commencing at orbit 884, right SAD motor winding volts increased to a maximum at orbit 893 of 8.11 V; subsequently, it has repeated these cycles | Unknown | Investigation indicated no immediate problem based on life test unit; performance not considered anomalous | Yes | None |
| 21 | Commencing in orbit 904, several occasions observed where IRLS-DCS interrogations were of 6 to 7 sec duration (normal is 3.8); during these extended interrogations, there was no entry into memory | Problem apparently due to the inherent logic design of the time code generator (internal timer) | Not considered anomaly by manufacturer since circuit performing as designed | No | None |
| 22 | During orbit 946, a short DCS interrogation (0.4 sec) occurred | Same as item 21 | Same as item 21 | No | None |
| 23 | During orbit 1153, optical end-of-tape switch at record end of HDRSS B failed; it subsequently failed 7 out of the next 17 times | Undetermined | Last failure was in orbit 1314; has worked properly since | Yes | Commands are stored during blind orbits to prevent recorder from going to the mechanical end-of-tape (only during blinds since that is only record interval long enough to utilize entire tape) |
| 24 | Irregularities in the IMCC earth scan, starting in orbit 1483 | Believed due to magnetization of bearing | Changes in scan occur, in most cases, 6 to 8 min after earth day, at about the time IMCC housing reaches its minimum temp. | No | None |

## Table C. 28-2. On-Orbit Anomaly Summary - Nimbus 4 (Cont.)

| No. | Anomaly | Cause | Remarks | Nimbus E Related | Correction Action |
|---|---|---|---|---|---|
| 25 | IRLS transmitter shut off after 46 sec; second try, it shut down after 146 seconds; third try normal | Believed to be caused by perturbation in data to IRLS (RF link noise) | None | No | None |
| 26 | SIRS channel 1 output has been saturated nearly every orbit since ≈ orbit 1600 | (Believed to be same as item 10) | Data excluded from NOAA operational program | Yes | None |
| 27 | IRLS - Short read-out (30 sec) | Unknown | A second DRO during the same interrogation 3645A was normal | No | None |
| 28 | Excessive noise bursts on HDRSS A data since orbit 3703 made data essentially unusable | Frequency of noise bursts corresponds to once-around of a bearing on the planetary shaft; bearing believed to be cause of the noise, cause now believed to have been debris in the bearing | Record time was decreased to 20-min, then 15-min intervals with good data; since orbit 4628, HDRSS A has been providing good quality data for ≦ 125 min of record | Yes | None |
| 29 | THIR radiometer scan motor failed during orbit 3731 | Believed due to debris in the gear assembly | THIR successfully restarted during orbit 3983 and has been rotating since | Yes | Various attempts made at cycling "on"/"off" commands of various duration; the final successful interval was after ≈20 orbits off and then successive series of on/off commands |
| 30 | Attitude perturbations occurred in pitch, roll, and yaw | Believed to be due to a missing earth pulse from the forward scanner followed by a full-on earth pulse of 28 to 30 sec | Control computation resumed within 30 sec | Yes | None |
| 31 | A command 306 in cell 5 of the redundant comstor changed to a 106 106 | Unknown | Comstor was loaded during orbit 4519 and operated successfully until 4560 | Yes | None; tests are set-up to periodically test cell 5 of the redundant comstor |
| 32 | SCR channels 5 and 6 signal outputs became unusable at orbit 4618 | Believed caused by a failure of the common converter of the bolometer bias supply 3 | | Yes | Various "on"/"off" attempts of various intervals unsuccessful; channels returned to normal by orbit 5022 |
| 33 | BUV photometer shutters did not return to the normal mode following calibration on 4 occasions; this was confirmed by the high voltage | Unknown | Under Investigation | No | BUV calibration is presently inhibited; it is planned to enable calibration once each week for 24 hr |

Table C.28-2. On-Orbit Anomaly Summary - Nimbus 4 (Cont.)

| No. | Anomaly | Cause | Remarks | Nimbus E Related | Corrective Action |
|---|---|---|---|---|---|
| 34 | RMP gyro failed during orbit 4905 | Believed to be mechanical, probably bearing seizure | Attempts to re-start unsuccessful | Yes | Changed yaw mode to sun sensor |
| 35 | THIR scan mirror motor failed at orbit 4973 | Results of the Failure Review Committee are "cause is felt to be due to a lubrication failure in one or more of the bearings and/or gear meshes" | This unit initially failed at orbit 3731; was successfully restarted in orbit 3983 and ran until orbit 4973 | Yes | None |
| 36 | HDRSS A failed to playback during orbit 5029 | Believed to be either broken negator spring or jammed planetary drive | None | Yes | None |
| 37 | Since orbit 5100 problems have existed with SIRS channe's 10, 12 and 13 | Believed to be the same as item 5 | None | Yes | None |

Table C.28-3. On-Orbit Anomaly Summary - Nimbus 5

| No. | Anomaly | Cause | Remarks | Nimbus F Related | Corrective Action |
|---|---|---|---|---|---|
| 1 | Solar array volts and unregulated bus volts rose erratically at the SN/SD transition +65 minutes; solar array 1 dropped slightly | No anomaly caused by shifting on the IV curve | Determined to be normal - not an anomaly | No | None required |
| 2 | ESMR preceased data shows "banding" through middle of picture | Characteristics of this type of antenna | Revision to EIS processing program has reduced problem | Yes | Modified EIS program |
| 3 | SCMR cone wall temperature higher than expected (292°K) | Not really an anomaly; concern was that temperature would exceed telemetry resolution of 296°K) | If temperature increases to 296°K, cone wall heater will be turned off | No | See remarks |
| 4 | ITPR scan motor temperature too high (52°C) | Presently unknown; believed due to increased torque requirement of scan motor | Expected temperature was 40°C; scan seemed proper until orbit 50 when became erratic | Yes | None |
| 5 | SCMR scan mirror speed telemetry decreased to zero during orbit 2 for 25 min; data during these periods unusable | Attributed to failure of the pip generator which appears to be temperature sensitive; pickup is made of several materials having different temperature coefficients of expansion by a factor of 2 to 4 | No data received from SCMR since orbit 320 | No | Operation initially resumed by increasing temperature of motor; however, temperature required soon exceeded capability of supply |
| 6 | ITPR scan erratic (orbit 50) | Not known; possibilities are:<br>• Scan motor torque marginal<br>• Required torque increased<br>• Possible contamination of lubricant used in harmonic drive | Limited scan attempted with limited success | Yes | Unit operated in nadir view most of the time |
| 7 | SCR high gain of channel D3 decreased to the point where earth, blackbody, and and space view levels remained at the clamp values (≈ orbit 201 - 210) | Found to be caused by "moonshine" and is not anomalous | Phenomenon repeated at ≈ orbit 600 | No | None required |
| 8 | SCR - Calibration mirror hung up in space view during a cal cycle | During cal cycle a normal gain command executed and this is believed to have caused an upset of logic | Suspected cause was concurred with by experimentors | No | None required |
| 9 | SCR - FOVC bit errors increased from approximately 4 to approximately 9 per orbit | Believed thermally related - increase in errors occurred in orbits where batteries were being overcharged to increase spacecraft temperature for stimulation of SCMR mirror motor | Error average returned to normal when spacecraft temperature returned to normal | No | None required |

Table C.28-3. On-Orbit Anomaly Summary - Nimbus 5 (Cont.)

| No. | Anomaly | Cause | Remarks | Nimbus F Related | Corrective Action |
|---|---|---|---|---|---|
| 10 | SCMR - Processed photographs of visible data defocused | Unknown, being investigated | Multiple imaging present in visible data for high contrast and independent of special offset and gain adjustment in ground processing equipment | No | |
| 11 | SCMR - Blackbody temps. for channels 1 and 2 vary in the video but not in telemetry | Unknown | Channel 1 variation more severe than channel 2 | No | |
| 12 | ESMR - Dropouts (2-60 sec) occur in digital A data | The ESMR Failure Review Committee believes problem in analog multiplexor integrate & dump filter circuit | Dropouts started at orbit 817 and did not occur every orbit but sometimes occurred 2 to 3 times in one orbit; after orbit 1063, reduced level has existed every orbit | Yes | |
| 13 | HDRSS A - Increase in flutter | Believed to be caused by bearing on the planetary shaft | Noise frequency is 106 Hz; this is similar to the problem observed on Nimbus 4 HDRSS A | Yes | Use of HDRSS A restricted to blinds |
| 14 | SCR - Field of view comp mirror errors increased from 9/orbit to 300-600/orbit | Attributed to a change in temperature; this telemetry point is temperature sensitive | Mirror motion believed to be OK | | None required |
| 15 | Clock - Both prime and redundant comstor execute times slipped by 4 min | During int. 1918 R several command problems experienced due to DTS link problems; subsequently noted that both comstors commands were executing ~4 min late | This same problem had occurred earlier on Nimbus 4 under the same conditions --DTS command problems | Yes | |
| 16 | SCMR - Visual channel sun cal output is 0.6 V versus expected 3.6 V | Possibly due to misalignment | | No | None |
| 17 | ITPR - Scan mirror went clockwise through 2 electrical stops when commanded to nadir at orbit 1600 | | Believed to be an encoder problem | Yes | |
| 18 | ITPR - Dig. WD 3 did not indicate any status change during retrace in grid 3; scan mirror missed a stop and subsequent stops were skewed | | Same as 17 | Yes | |

## Table C.28-3. On-Orbit Anomaly Summary - Nimbus 5 (Cont.)

| No. | Anomaly | Cause | Remarks | Nimbus F Related | Corrective Action |
|---|---|---|---|---|---|
| 19 | ITPR - Logic/encoder problems — scan failed to locate space position | | Same as 17 | Yes | |
| 20 | SCR - Gain of "A" channels decreased by approximately 2-1/2% and noise increased between orbits 1660 to 1680 | Detector suspected | | Yes | |
| 21 | SCMR - Visual channel sun cal. amplitude is 0.6 V versus expected 3.6 V | Believed due to optical misalignment | | Yes | |
| 22 | VIP - Beacon signal fades/dropouts at Alaska during orbits with ascending mode between 20°W and 70°W and 110° to 120°W | Believed due to a hole in the radiation pattern caused by combination of deployed ESMR antenna plus SAD position of 140° to 160° | Fade/dropouts duration usually less than 2 min and presents no problem | No | |
| 23 | SCMR tape recorder ±12 supplies do not reach proper level for approximately 32 sec | | | No | |
| 24 | SCR - Cal volt fcn #15921 out-of-limits 550 times out of 6000 samples | Unknown | None | Yes | |
| 25 | ESMR - Data level slightly reduced | Believed to be different manifestation of the original problem | None | Yes | |
| 26 | Power - A notch occurs in the solar array current (≈900 mA) near EN/ED transition nearly every orbit | Not known | Notch duration varies from 0 to 300 sec | Yes | |
| 27 | Power/ACS/ISM-telemetry power to 10 functions on the SADs lost due to apparent blown fuse | Reason for fuse clearing not known | Fuse cleared at time of notch; telemetry circuits get power now during SD from paddle volt telemetry circuit | Yes | |
| 28 | Clock - HDRSS A "off" command 734 failed to execute | Unknown | Execution counter increased | Yes | |
| 29 | ACS - Pitch, roll, and yaw attitude perturbations | Caused by a missing earth pulse from forward | Has not been observed since or previously | Yes | |
| 30 | SCR - Gain decrease on channels A, C, and D | Unknown | Change was over a long period time | No | |
| 31 | Clock - Prime comstor went to an improper (unknown) state when the verify command sent to the redundant comstor | Unknown | Both comstors tested and are OK | Yes | |

## C.29 SYNCHRONOUS METEOROLOGICAL SATELLITE (SMS)

### C.29.1 Program Summary

The SMS program was initiated by NASA to provide a geostationary weather watch capability. The satellites are built for NASA at the Philco-Ford Corporation Western Development Laboratories (WDL) facility in Palo Alto, California.

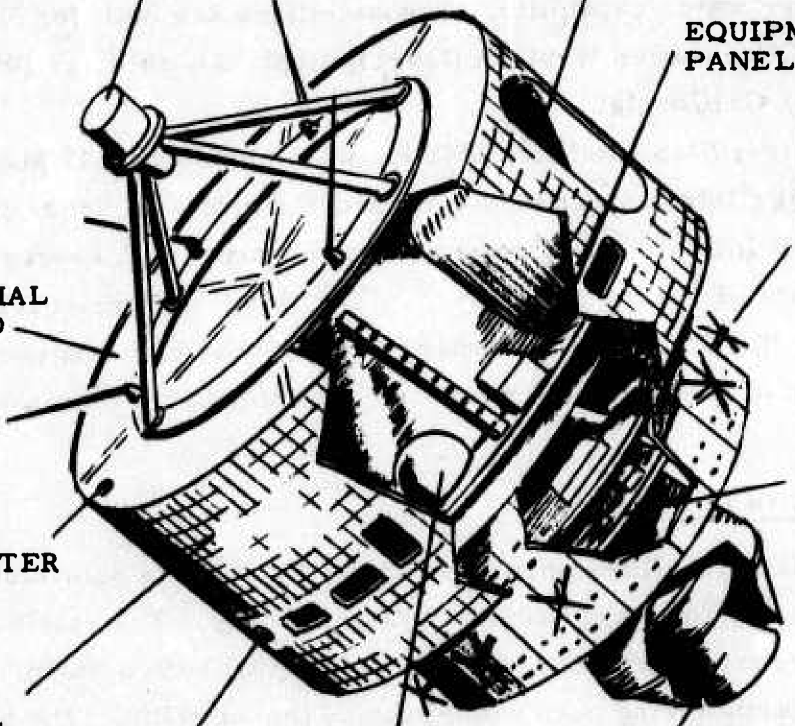
The first SMS satellite, SMS A, was launched on 17 May 1974 aboard a Delta booster into a synchronous equatorial orbit. Another Delta booster placed SMS B into a similar geosynchronous orbit on 6 February 1975. A third launch is planned for the fall of 1975. The third and subsequent spacecraft will have the GOES (Geostationary Operational Environmental Satellite) designation of the National Oceanic and Atmospheric Administration (NOAA).

### C.29.2 Satellite Description

The SMS satellite, shown in Figure C.29-1, is spin-stabilized at 100 rpm about the axis of its cylindrically shaped body. The satellite contains a visible infrared spin scan radiometer (VISSR) with a 16-in. aperture that scans the earth during each revolution of the satellite. The solar array is body-mounted with openings for attitude control earth horizon sensors; other sensors are mounted on the equipment platform.

Attached to the main body of the satellite is a smaller cylinder that supports the multiple elements of both S-band and UHF antennas. In operation, the antenna beam is electrically despun in the opposite direction and at the same rate as the spinning satellite so that the beam remains pointed toward the earth.

A tripod at the forward end of the satellite serves as the support for a magnetometer. Adjacent to the tripod are four monopole antennas used for the VHF telemetry and command subsystem.




**LAUNCH DATES**

| | | |
|---|---|---|
| SMS-A | - | 17 MAY 1974 |
| SMS-B | - | 6 FEB 1975 |
| BOOSTER | - | DELTA 2914 |
| ORBIT | - | SYNC. EQUATORIAL |

**CHARACTERISTICS**

| | | |
|---|---|---|
| DIAMETER (MAX) | - | 6 FT |
| HT. (OVERALL) | - | 11.3 FT |
| WT. (LIFTOFF) | - | 1379 LB |
| POWER (BOL) | - | 200 WATTS |
| DESIGN LIFE | - | 5 YEARS |

Figure C.29-1. Synchronous Meteorological Satellite (SMS)

All SMS subsystems were designed for a satellite lifetime goal of 5 years. A summary of the major subsystems is given below. Detailed subsystem descriptions and other technical data can be found in References C.29-1 through C.29-4.

### C.29.2.1 Power Subsystem

The power subsystem is made up of three solar arrays, two batteries, and a power control unit (PCU). These components together supply all satellite power by solar array output and/or from storage in the batteries.

The solar arrays provide a minimum power (equinox) of 212 W at beginning of life and 157 W (solstice) at end of life. Each of the two redundant batteries has a capacity of 3 A-hr.

### C.29.2.2 Attitude Determination and Control (ADAC) Subsystem

The ADAC is composed of five major components: earth sensor assemblies, sun sensor assembly, nutation sensor assembly, ADAC electronics unit, and passive nutation damper. The subsystem provides the following functions:

a. Spin axis orientation determination
b. Spin rate determination
c. Angle-referenced timing signals
d. Antenna control
e. Nutation sensing and active nutation damping control signals
f. Passive nutation damping

The ADAC includes all the signal conditioning electronics necessary to interface with the dual telemetry unit, the VISSR DM, the VISSR, the SEM components, and the Comm subsystem.

### C.29.2.3 Telemetry and Command (T&C) Subsystem

This subsystem is made up of a dual telemetry unit, a command unit, a VHF transponder, a VHF diplexer, and a VHF antenna. These components provide the following functions:

a. Pulse code modulation (PCM) and real-time telemetry for monitoring satellite status

b. Real-time commanding capability (193 total commands)

c. Ranging capability for use at VHF during launch, synchronous orbit acquisition, and eclipse periods

The command unit and dual telemetry unit are also used together with the S-band equipment during synchronous orbit to provide S-band telemetry for satellite status monitoring and S-band commanding capability. The VHF transponder, command unit, and dual telemetry unit are largely redundant equipment.

### C.29.2.4 Communication (Comm) Subsystem

The Comm subsystem consists of redundant S-band transponders, each of which include one S-band receiver and one S-band transmitter, redundant UHF transponders, coaxial and power switches, the UHF power amplifier control unit (PACU), and the communication antenna assembly. Together, these components perform the following functions:

a. Transmit VISSR wideband at S-band to the NOAA Command and Data Acquisition (CDA) station at Wallops Island.

b. Receive, at S-band, stretched VISSR data, weather facsimile (WEFAX), or trilateration ranging signals, as well as data collection platform interrogation (DCPI); translate in frequency; amplify and retransmit, at S-band, the stretched VISSR data to Regional Data Utilization Stations (DUS), WEFAX to Automatic Picture Transmission (APT) stations, and ranging signals to trilateration stations; retransmit the DCPI at UHF.

c. Receive data collection platform reports (DCPR) at UHF and retransmit at S-band to the CDA station.

d. Accept, at S-band, command signals and condition for processing by the command equipment and transmit, at S-band, telemetry data provided by the telemetry equipment.

C.29.2.5 VISSR Digital Multiplexer (VISSR DM)

This unit accepts the eight visible and two infrared outputs from the VISSR and processes them for quadriphase modulation (QPSK) of the S-band transponder. They are analog/digital (A/D) converted, multiplexed, formatted, and the resulting digital bit stream quaternary encoded to condition it for the quadriphase modulator in the S-band transponder. The DM has redundant channels for the VISSR data.

C.29.2.6 Space Environment Monitor (SEM) Subsystem

The SEM subsystem is composed of three separate major instruments: the magnetometer, the energetic particle sensor (EPS), and the x-ray instrumentation. The magnetometer monitors the magnitude and direction of the magnetic field; the x-ray instrumentation monitors the intensity of solar x-ray radiation; and the EPS monitors the energy level and quantity of energetic particles.

The instruments include all the data processing and signal conditioning electronics necessary to interface with the satellite telemetry equipment.

C.29.2.7 Auxiliary Propulsion Subsystem (APS)

The APS is an all-hydrazine subsystem that operates from an unregulated (blowdown) direct-pressurization propellant supply. The APS contains six thrusters, arranged to afford back-up capabilities for every propulsive maneuver. The six-thruster complement is composed of two pairs of 5-lb thrusters and one pair of 0.5-lb thrusters. All thrusters are fed from a common supply manifold.

Three identically sized propellant storage tanks are interconnected on the gas and liquid sides. The liquid manifold contains a fill-drain valve. The feed system is configured to permit ground testing and servicing operations such as filling, pressurizing, draining, and venting, as well as to satisfy essential flight operational requirements of propellant delivery, mass distribution, and phase separation.

C.29.2.8 Apogee Boost Motor (ABM) and ABM Adapter

The ABM consists of a solid rocket motor chamber with solid propellant, redundant initiator system with safe and arm device, and an aft closure and expansion nozzle. The use of the ABM is to impart sufficient velocity to inject the satellite into a circular equatorial orbit at apogee.

C.29.2.9 Satellite Separation Equipment

The separation equipment consists of components used to separate the ABM/ABM adapter from the satellite and the VISSR cooler cover and cooler cover release mechanism.

The VISSR cooler cover makes a dust-tight fit against the VISSR radiation cooler and provides thermal protection to the cooler during ABM heat soakback. The cover is ejected from the satellite by ground command following ABM separation.

C.29.2.10 Structure

The structure consists of a central shell supporting horizontal equipment platforms, the ABM, and the VISSR. The lower end of the shell interfaces with the launch vehicle. The lower portion of the structure (ABM adapter) and the ABM case are jettisoned after ABM burn to provide a stable inertia ratio and to provide a clear field of view for the VISSR radiation cooler. The structure incorporates appropriate framework for mounting and support of the various components and subsystems.

C.29.2.11 Thermal Control Subsystem

The thermal control subsystem consists of thermal insulation assemblies, shield assemblies, and heaters required to maintain the satellite and its components within their respective temperature limits. The satellite thermal control is achieved passively, with the exception of heaters on the APS tanks, lines, and thrusters (to ensure that the APS propellant does not freeze and to ensure thruster performance), magnetometer sensor, and radiation cooler.

C.29.3 Reliability and Testing

Details of the various phases of SMS reliability assessment and testing through launch are given in References C.29-5 to C.29-22.

C.29.4 Key Events and Milestones

Significant milestones in the SMS program are shown in Figure C.29-2. Major failures and malfunctions are also indicated.

C.29.5 Malfunction/Anomaly Summary

C.29.5.1 SMS A

C.29.5.1.1 Launch Base

After a 1-day delay (due to booster hydraulic problems), SMS A was launched on 17 May 1974 at 5:31 a.m. EDT. A malfunction in the first stage resulted in a transfer orbit apogee that was too low by 345.4 km (a 6° error) and a 9° error in inclination angle.

The launch error was overcome by using the auxiliary propulsion system (APS) fuel for orbital corrections, and the spacecraft was finally positioned at 45°W longitude.

C.29.5.1.2 On-Orbit

17 May - 17 July 1974

a. Ground command switched to side 2 ADAC based on very low nutation level telemetry indicating failed side 1. Actual tapes confirmed side 1 nominal.

b. A premature indication of ABM separation occurred during ABM firing.

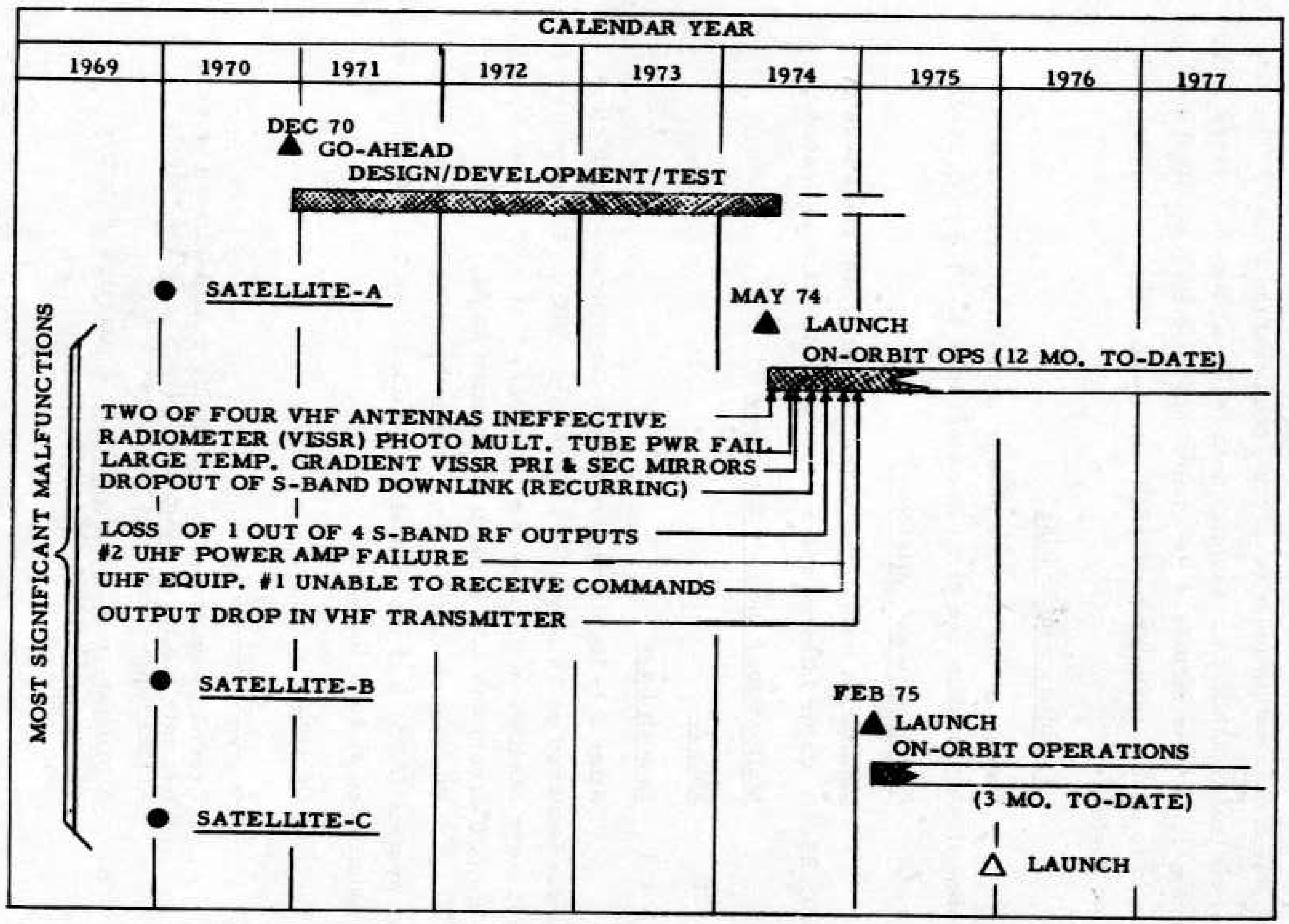



Figure C. 29-2. Key Milestones and Events - Synchronous Meteorological Satellite (SMS)

c. ADAC side 2 spin clock telemetry intermittent when using earth sensor reference mode. Loss of S-band signal coincided with some spin clock errors. Lunar conflict occurs when using earth sensor reference - about twice a month. These conflicts can be successfully predicted and avoided by reselecting the earth sensor for timing.

d. Low level x-ray signal goes negative - studies indicate x-ray sensor is susceptible to ultraviolet radiation, which affects the low level resolution of instrument.

e. VISSR stair-step calibration sequence appears randomly at beginning/end or during picture.

f. VISSR mirror step current drops off during picture. Most likely, this is the result of less torque required to move mirror after bearings are rotated away from their normal rest position in their races.

g. S-band side 2 RF output problem; equipment 1 selected and turned on.

h. S-band side 1 power amplifier temperature higher than expected; temperature safety margin found to be adequate.

i. VISSR IR response 20 percent low (reduced later to 7 percent).

j. VISSR channel 7 photomultiplier tube (PMT) power supply failed. Coincident with this failure was a 1°C rise in the temperature of both calibration shutters and the primary mirror.

k. A large temperature gradient exists between the primary and secondary VISSR mirrors. This gradient is larger than expected, and it has resulted in a out-of-focus condition in the visible channels.

l. Energetic particle sensor telemetry indicates that EPS calibration sequence does not start immediately upon command, but rather is delayed until the start of a new telemetry subframe cycle.

m. Telemetry subsystem marked by two minor problems:

   1. Fuel pressure display interference when UHF power amplifier is on. Data are usable and considered accurate.

   2. UHF power amplifier 2 temperature telemetry inoperative (probable sensor failure).

n. Spacecraft telemetry indicated UHF transmitted power down 3 dB on day 153. Test results, however, indicated equipment operation nominal.

o. A problem developed shortly after launch with the VHF antenna network. A cable or connector is loose which effectively results in only two of the four antennas being active. This produces an antenna pattern with large nulls, up to 18 dB ripple. This problem has been corrected on the SMS B and C spacecraft.

<u>17 July - 18 August 1974</u>

a. Several dropouts in the S-band downlink signal have occurred. They are about 15 dBm in magnitude and several seconds in duration. At least six dropouts have been recorded. Possible cause may be that the S-band antenna (electrically despun) is "swinging off" the earth.

<u>18 August - 20 September 1974</u>

a. The response characteristics of the individual visible sensors have diverged, resulting in a "stripping" effect in the VISSR pictures.

b. A white (cold) streak occurs across the infrared pictures whenever the moon is about to enter the VISSR field of view. This phenomenon can occur on three or four pictures a month.

c. A large amount of visible light scattering occurs whenever the sun appears in the VISSR aperture.

d. The VISSR PMT power supplies exhibit a temporary reaction to the high brightness level associated with the sun and the surrounding scatter halo. The normal manifestation is an instantaneous 10- to 200-V drop in PMT power supply voltage as a visible channel scans the sun.

e. The AFT solar array temperature sensor failed during eclipse on 14 September. The loss of the sensor has no operational significance.

f. On day 153, spacecraft telemetry indicated the UHF transmitted power was down by 3 dB. On day 262 (September 19), the UHF system experienced a second anomaly where both telemetry and received signal indicated no output. The system returned to nominal operating condition.

g. The loss of output from two of the four VHF antenna elements recurs daily, for an average of 10 minutes, during the extremes of the eclipse temperature excursions.

<u>20 September - 20 October 1974</u>

a. The VISSR DM 10-V reference has been observed to vary in proportion to the amount of visible light entering the VISSR visible channel field of view.

b. The ADAC lost earth lock on 28 September. All events pointed to a moon conflict on the earth sensor. However, it is possible this anomaly is related to the occasional S-band dropouts.

c. On 14 October, one of the four S-band RF outputs failed, the power dropping to zero, but the S-band link remained usable.

d. On 17 October, UHF power amplifier of equipment 2 failed. UHF equipment 1 was switched on but failed to receive the uplink signal. The failure of equipment 1 appears to be due to a cable connector or solder joint. Equipment 2 has "listen only" capability and equipment 1 has "transmit only" capability. Equipment 1 and 2 receive and transmit functions cannot be paired up because they are not cross-strapped.

<u>21 October - 20 December 1974</u>

a. Telemetry unit 1 was selected on 5 December 1974 (normal switch over). Some analog points are reading 1 PCM bit higher than on telemetry unit 2.

b. Total S-band RF output dropped approximately 3 W on 17 December. Total RF output power eventually returned to 15.1 W. (Note: Only three of the four power amplifiers are operating.)

c. VHF transmitter 1 telemetered high power mode value decreased from 8.48 W to 7.39 W (confirmed by minor signal degradation).

<u>12 December - 20 January 1974</u>

No new anomalies reported.

C.29.5.2 SMS B

C.29.5.2.1 Launch Base

SMS B was successfully launched into a synchronous orbit on 6 February 1975.

C.29.5.2.2 On-Orbit

Preliminary data indicate that all primary and secondary mission functions are being successfully performed.

## References


C.29-1. Synchronous Meteorological Satellite Phase C Design Report, WDL-TR-4545, Philco-Ford Corporation, Western Development Laboratories, Palo Alto, Calif. (30 June 1971).

C.29-2. Satellite Test Plan for Synchronous Meteorological Satellite (SMS), Specification No. SA 212117, Rev. A (5 January 1973).

C.29-3. Geostationary Operational Environmental Satellite (GOES)/ Synchronous Meteorological Satellite (SMS) Operations Manual, National Aeronautics and Space Administration, Goddard Space Flight Center, Greenbelt, Md.

C.29-4. GOES Baseline Power Profile Report, DR 010, GOES-PCC-117 (10 January 1975).

C.29-5. Vibration Test Report of the Development Model SMS, SMS PCC 4415 (2 May 1972).

C.29-6. Final Report of SMS Static Load Test, SMS-PCC-4125.

C.29-7. SMS Development Shock Test Report, SMS-PCC-4468 (18 May 1972).

C.29-8. SMS Spacecraft-Thermal Development Test Report, SMS-PCC-4320 (10 April 1972).

C.29-9. Launch Readiness Review for Synchronous Meteorological Satellite SMS A (23-24 April 1974).

C.29-10. Launch Readiness Review for Synchronous Meteorological Satellite SMS B (18-19 December 1974).

C.29-11. Thermal Vacuum Chamber Operations During Satellite Qualification Tests, SMS Test Procedure SC 236197.

C.29-12. Preparation for and Qualification Vibration Tests, SMS Test Procedure SC 236193.

C.29-13. Thermal Vacuum Chamber Operations During Satellite Acceptance Tests, SMS Test Procedure SC 236223.

C.29-14. Preparation for and Acceptance Vibration Tests, SMS Test Procedure SC 236220.

C.29-15. *Synchronous Meteorological Satellite Launch Base Test Plan*, WDL-TR-4796, Philco-Ford Corporation, Western Development Laboratories, Palo Alto, Calif. (8 February 1974).

C.29-16. *Synchronous Meteorological Satellite Development Test Report*, WDL-TR-5021, Philco-Ford Corporation, Western Development Laboratories, Palo Alto, Calif. (30 June 1972, Vol. I; 1 December 1972, Vol. II).

C.29-17. SMS-PCC-5320, *SMS Final Reliability Assessment* (17 November 1972).

C.29-18. *SMS Quantitative Reliability Assessment Specification*, WDL-ST-A20403, Philco-Ford Corporation, Western Development Laboratories, Palo Alto, Calif. (5 March 1971).

C.29-19. *SMS Reliability Program Plan*, WDL-TR-4492 (and Appendices), Philco-Ford Corporation, Western Development Laboratories, Palo Alto, Calif. (February 1971).

C.29-20. *Preferred Parts List*, GSFC-PPL-12, National Aeronautics and Space Administration, Goddard Space Flight Center, Greenbelt, Md. (July 1972).

C.29-21. *Reliability Predictions*, MIL-HDBK-217A (1 December 1965).

C.29-22. *Preferred Parts List*, GSFC-PPL-11, National Aeronautics and Space Administration, Goddard Space Flight Center, Greenbelt, Md.

## C. 30 IMPROVED TIROS OPERATIONAL WEATHER SATELLITE (ITOS)

### C. 30. 1 Program Summary

NASA's ITOS operational weather satellites, built by RCA, provide both visible and infrared meteorological data for use in weather predictions and meteorological studies by the National Oceanic and Atmospheric Administration (NOAA). The satellites also provide secondary data on earth heat balance and solar proton and electron flux. Local data are transmitted in real time to individual APT (automatic picture transmission) stations located around the earth, and stored global data are transmitted to the Command and Data Acquisition (CDA) stations of the ITOS ground network.

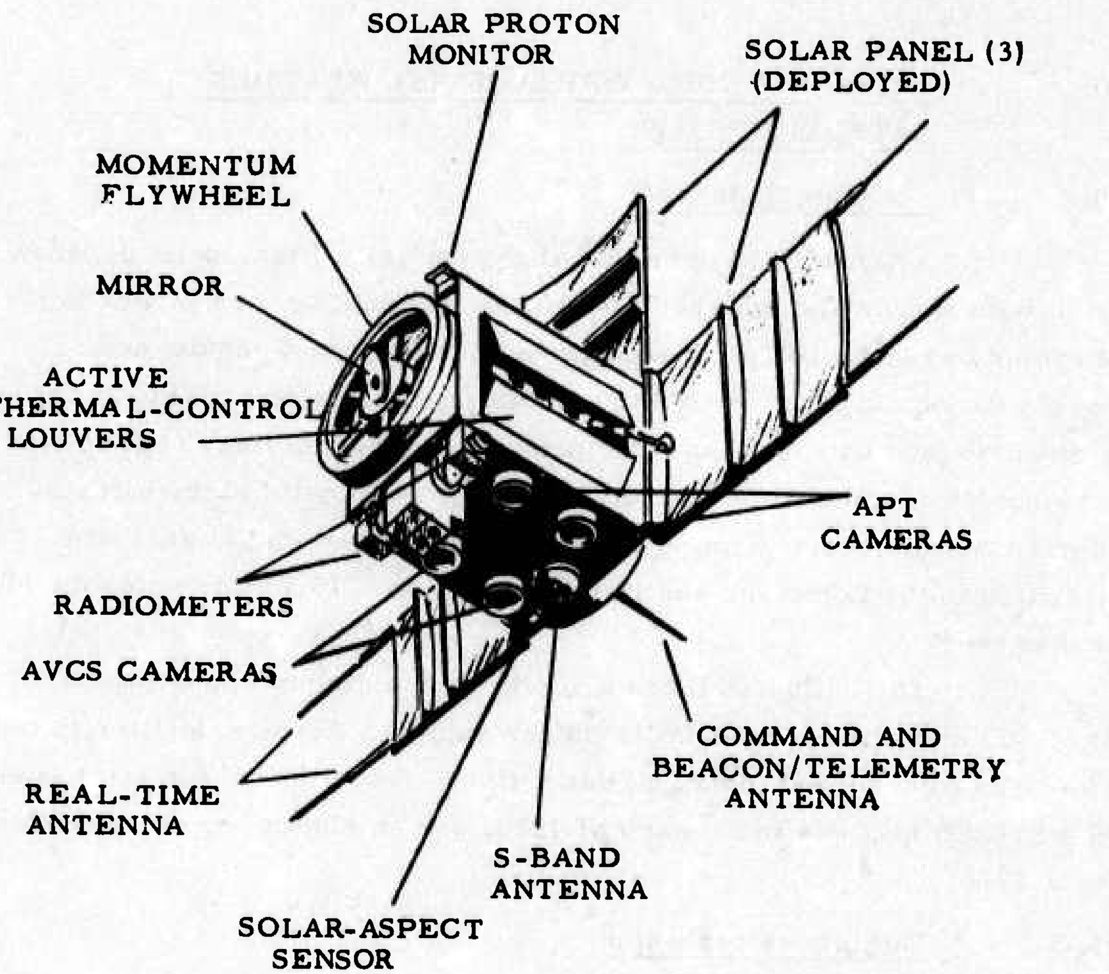
The ITOS satellites are placed in a circular, near-polar orbit of 790 nmi altitude by a Delta launch vehicle. Seven satellites in the ITOS series have been launched to date, five successfully. The first launch, ITOS 1 (TIROS M), was in January of 1970, and an eighth launch is scheduled for late 1975.

### C. 30.2 Satellite Description

The general characteristics of the ITOS satellites are shown in Figure C. 30-1. Additional details of the TIROS M and ITOS D series spacecraft are given below. The ITOS D is an extension of the basic TIROS M/ITOS system.

#### C. 30.2.1 TIROS M/ITOS Series

The TIROS M spacecraft (ITOS 1 satellite) is a rectangular prism with a deployable solar array and a momentum wheel that provides gyroscopic stabilization of spacecraft earth-oriented attitude. Reaction between the earth's magnetic field and a magnetic field generated within the spacecraft provides control over the yaw and roll components of spacecraft attitude and the total system momentum. A sampled data servo system with pitch error input (derived from the attitude relationship between the




LAUNCH DATES

| | | |
|---|---|---|
| ITOS 1 | - | 23 JAN 1970 |
| NOAA 1 | - | 11 DEC 1970 |
| ITOS B | - | 21 OCT 1971 |
| NOAA 2 | - | 15 OCT 1972 |
| ITOS E | - | 16 JULY 1973 |
| NOAA 3 | - | 6 NOV 1973 |
| NOAA 4 | - | 15 NOV 1974 |

BOOSTER - DELTA 2310

ORBIT - 790 NM
$97^{\circ}$ INCLINATION

CHARACTERISTICS

| | | |
|---|---|---|
| WIDTH (ASCENT) | - | 40 IN |
| WIDTH (DEPLOYED) | - | 258 IN |
| HT (OVERALL) | - | 75 IN |
| WT (LIFTOFF) | - | 750 LB |
| POWER (BOL) | - | 350 WATTS |
| DESIGN LIFE | - | 2 YEARS |

Figure C. 30-1. Improved TIROS Operational Weather Satellite (ITOS) spacecraft and the earth's horizon) provides the required control of spacecraft pitch motion by means of momentum interchange between the wheel and spacecraft.

Thermal control of the spacecraft is provided by both active and passive techniques. The passive technique is implemented by a thermal fence and appropriately distributed radiator surfaces. The thermal fence provides a variable absorptivity characteristic with respect to sun angle. The passive system is augmented by active thermal controllers (ATCs), which maintain temperature control by the variation of radiator surface area.

The power supply subsystem is a solar cell/battery/regulator system very similar to the TOS (TIROS Operational Satellite) system but with greater capacity; this subsystem will provide power for all operations of the spacecraft throughout its minimum mission life of 6 months. The primary environmental sensor subsystems provide the means for both direct readout and remote recording of cloud cover data during daytime and nighttime operation.

Direct readout of daytime cloud-cover picture data is performed by one of the redundant APT camera systems, which transmit picture data directly to APT ground stations (APTGS). Direct readout of nighttime infrared cloud-cover and cloud-top temperature data is accomplished by one of the redundant scanning radiometers (SR). The SR output is transmitted in real time to provide local nighttime coverage and also is stored aboard the spacecraft on redundant SR tape recorders to obtain global data that are played back and transmitted to one of the CDA ground stations. Daytime global cloud-cover data are provided by one of the redundant advanced vidicon camera subsystems (AVCS); AVCS picture data are stored on one of the redundant tape recorders. The stored daytime coverage information is later played back and transmitted from the spacecraft to a CDA station, from which it is relayed to the National Environmental Satellite Center (NESC). A back-up mode of operation is provided by the SR, which contains a visible channel in addition to the infrared channel. The visible channel can be used to provide visible daytime data for real-time transmission as well as a limited capability for global coverage with the SR recorder. Secondary sensors include a solar proton monitor (SPM) and flat plate radiometer (FPR).

The SR infrared channel "sees" radiance patterns rather than clouds as such and, as a result, interpretation of the infrared data in terms of cloud structures differs from the interpretation of the visual data. The two types of data support each other in providing two aspects of the same cloud structures.

The data generated by these subsystems are recorded on an incremental digital recorder and played back and transmitted to a CDA station. The CDA station relays the data to NESC for processing and analysis. The secondary data subsystem is not redundant. However, provision is made for back-up, real-time transmission of SPM data over the beacon and telemetry link.

The communications links between the satellite and ground stations include a beacon and telemetry link, a command link, and a real-time data link (APT and SR), all in the VHF band, and a playback data link (AVCS, SR, FPR, and SPM) in the S-band.

Redundancy, at both the component and the subsystem levels, is used extensively in the spacecraft design to ensure a high level of performance of spacecraft functions and to avoid the loss of any function as the result of single unit failure. Cross-coupling of components within a subsystem permits the selection of optimum signal-handling paths and may be used to circumvent multiple unit failures.

### C. 30.2.2 ITOS D Series

The ITOS D spacecraft, shown in Figure C. 30-1, is an adaptation of the TIROS M/ITOS spacecraft bus, utilizing essentially the same power subsystem, dynamics subsystem, thermal control subsystem, command subsystem, and communications subsystem to support the operation of a new complement of environmental sensors. The ground station equipment, also essentially the same as the TIROS M/ITOS equipment, and compatible with all previous TOS/ITOS/ESSA/NOAA spacecraft, has been modified to handle the ITOS D sensor data.

C. 30.2.2.1 Sensors

The primary environmental sensor subsystems provide the means for both direct readout and remote recording of earth cloud cover in the visible and infrared regions, and for remote recording of temperature profile data. The sensor subsystems are operated continuously to provide global coverage.

Direct readout of daytime and nighttime cloud cover data in both the visible and infrared regions is accomplished by operating both very high resolution radiometers (VHRR) in the phased-mirror mode. The time-multiplexed data are transmitted directly to VHRR ground stations (VHRRGS). Up to 9 minutes of time-multiplexed VHRR data may be stored on a VHRR recorder for subsequent transmission to a CDA station. Direct readout of both nighttime and daytime cloud-cover and cloup-top temperature data is accomplished by one of the redundant scanning radiometers (SR). The SR output, both in the visible and the infrared regions, is transmitted in real time and is stored aboard the spacecraft on redundant SR tape recorders to obtain global data that will be played back and transmitted to one of the CDA stations. Both the real-time and stored data will be transmitted in a time-multiplexed mode.

A vertical temperature profile of the atmosphere is obtained from the vertical temperature profile radiometer (VTPR) and recorded, on a global basis, on a track of the scanning radiometer recorder (SRR) for subsequent transmission to a CDA station.

The solar proton monitor (SPM) data are transmitted continuously in real time and also recorded globally, along with the VTPR data on a track of the SRR.

The CDA station relays the recorded SR, VHRR, telemetry, SPM and VTPR data to the NESC for processing and analysis. The real-time transmission of the SPM data is via the beacon telemetry link.

C. 30.2.2.2 Factors Affecting Design Life

The only expendable in ITOS D is the lubrication for the various bearings. The amount of lubricant included in the spacecraft is sufficient for several years under worst case conditions, so that the mission life requirement is easily met.

Wearout points in the system are tape recorders, SR bearings, VTPR filter motor bearings and gear drive, VHRR bearings, and the momentum wheel assembly (MWA) bearings. The VHRR, recorders, and MWA designs were conservative and no wearout during several years' operation is expected. None has been experienced.

There have been on-orbit failures of the SRs and VTPRs that were attributed to lubrication problems, but these occurred after the expected orbit life.

C. 30.2.2.3 Technological Advances

Since most of the components of ITOS D had been proven earlier on TIROS M (ITOS 1) and ITOS A (NOAA 1) in orbit, no unexpected innovations were necessary.

The MWA with a brushless motor was proven successful in orbit on ITOS D (NOAA 2). This change overcame the rapid brushwear problem experienced on ITOS 1 and NOAA 1.

C. 30.2.2.4 Design Margins and Compromises

The design margins established by analysis proved satisfactory. There were no design compromises on ITOS D caused by budget or schedule constraints.

C. 30.3 Key Events and Milestones

Major milestones of the ITOS program are shown in Figure C. 30-2. Significant on-orbit failures are also indicated.

C. 30.4 On-Orbit Malfunctions

Summaries of on-orbit malfunctions experienced in the ITOS program are given in Tables C. 30-1 through C. 30-5.

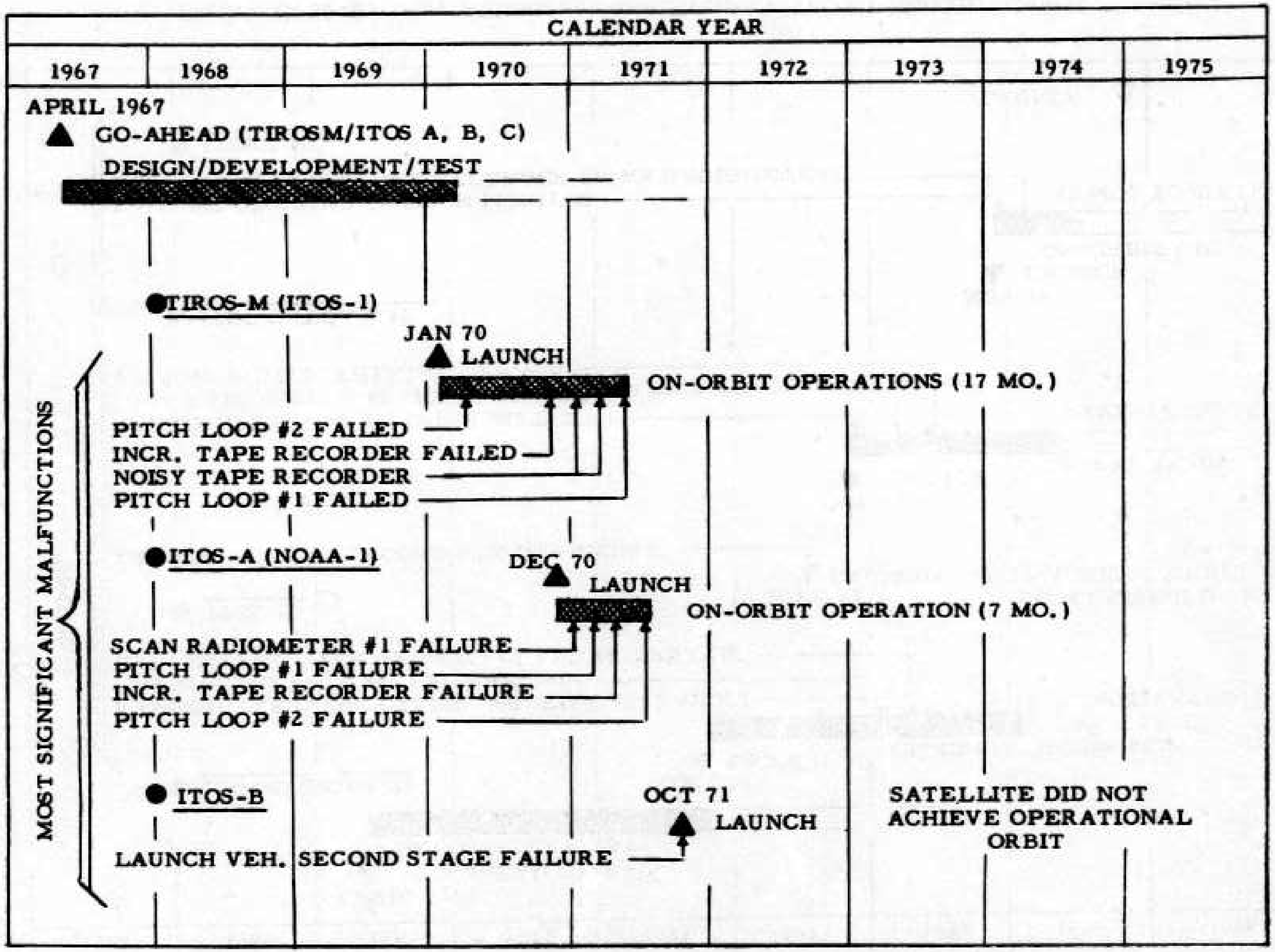



Figure C. 30-2. Key Milestones and Events - ITOS Program

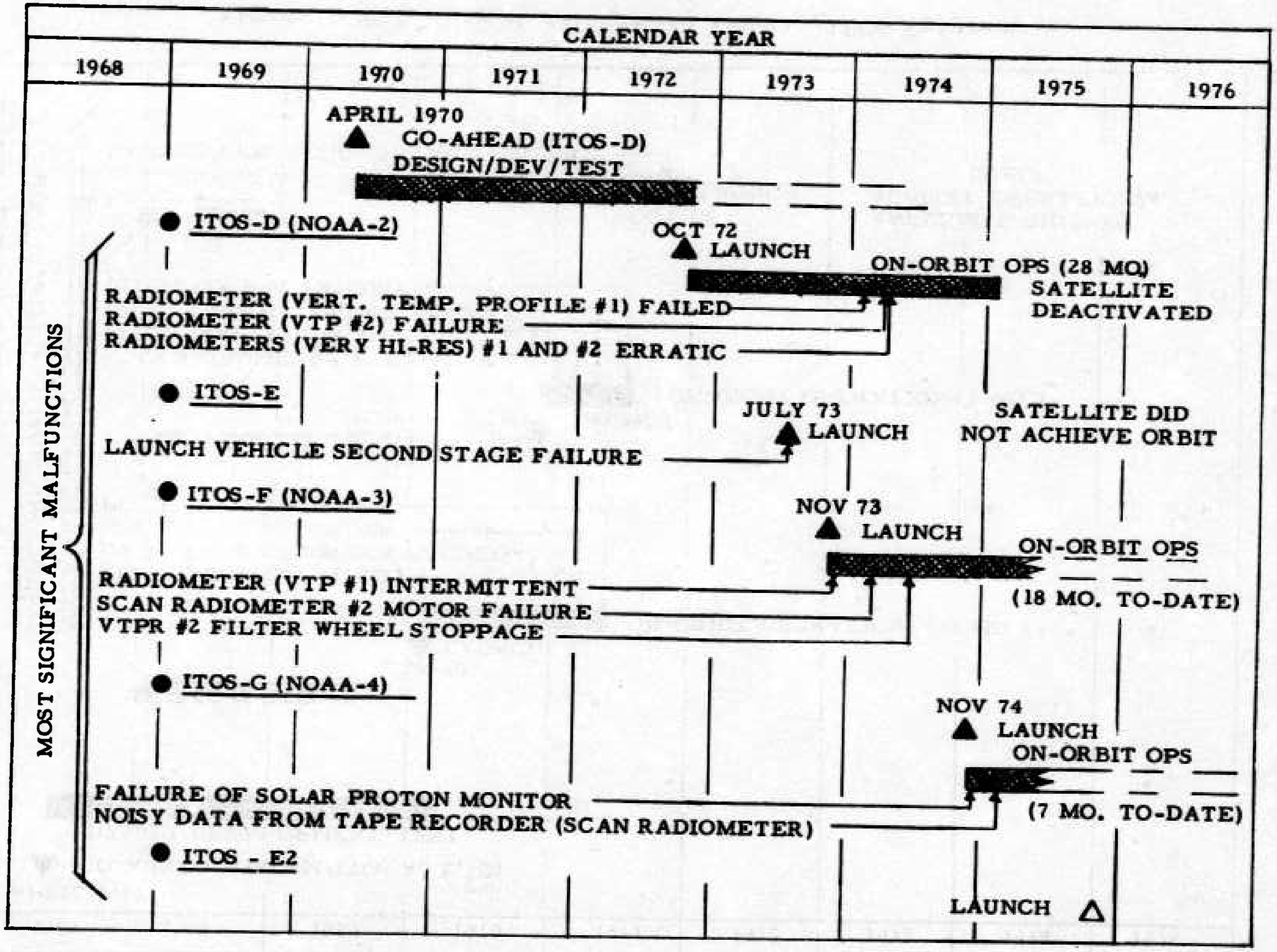



Figure C. 30-2. Key Milestones and Events - ITOS Program (Cont.)

Table C. 30-1. On-Orbit Malfunction Summary - TIROS M (ITOS 1). Launched 1/23/70

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 2/7/70 | MWA[a] motor #2 brush wearing at excessive rate | Probably due to low (+ 7°C) MWA temperature | None at this time | None |
| 3/1/70 Orbit 460 | MWA bearing temperature began rapid rise from + 18°C | Belived due to brush debris shorting motor commutator | None | None |
| 4/4/70 Orbit 893 | MWA bearing temperature reached max. temperature at + 44°C | | | |
| 4/5/70 Orbit 905 | MWA bearing temperature experienced sudden drop from + 44°C to + 16°C | Removal of brush debris from commutator due to brush having worn down to spring | None | None |
| 4/5/70 Orbit 905 | All brush reserve of MWA motor #2 consumed | Unknown | | |
| 4/5/70 Orbit 912 | Failure of pitch loop 2 | Excessive rate of brush wear and accumulated brush debris on the commutator | None - loss of redundancy in the pitch loop subsystem | None |
| 4/8/70 | MWA bearing temperature began an increasing trend from + 16°C | Believed to have been due to brush debris shorting between commutator segments | None | None |
| 4/13/70 | MWA bearing temperature began a rapid increase from + 33°C | Accumulated brush debris between commutator segments | None | None |
| 4/25/70 Orbit 1144 | Loss of pitch lock for about 13 min | Unknown, but suspected to be due to spring of worn out motor #2 brush binding commutator | Loss of picture coverage for 13 min | None |

[a]MWA - Momentum Wheel Assembly

Table C. 30-1. On-Orbit Malfunction Summary - TIROS M (ITOS 1). Launched 1/23/70 (Cont.)

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 11/17/70 Orbit 2107 | Incremental tape recorder failed | Stalled motor due to increased friction or slackening of tape causing capstan to slip | Failure occurred 5 months after mission requirements met | None |
| -- | Coherent noise in SR[b] data | Spacecraft tape recorder flutter and wow | Prevented calculation of sea surface temperatures to the desired 1°C accuracy | Design change made in SR recorder to accomplish Z axis correction on the spacecraft for ITOS D and later spacecraft; Z axis correction designed into the ground stations prior to ITOS D spacecraft launch |
| -- | Housekeeping telemetry commutators unintentionally stopped by commands to ESSA 9 satellite | Improper command address transmitted followed by a dummy word containing all zeros | Two frames of telemetry had to be run to get full set of data | CDA[c] station programmers were changed to transmit the 13th bit of the dummy word "enable" sequence to a "1" |
| 6/7/71 | Pitch loop #1 failed | Accumulation of brush debris on the motor #1 commutator | End of mission as pitch lock was lost more than 11 months after mission requirements were met | MWA changed to a brushless motor configuration on ITOS B and later spacecraft |

[b] SR - Scanning Radiometer

[c] CDA - Command and Data Acquisition (ground station)

Table C. 30-2. On Orbit Malfunction Summary - ITOS A (NOAA 1), Launched 12/11/70

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 1/1/71 Orbit 250 | MWA[a] motor #2 brush wear rate excessive (9 mils/20 revs) | Unknown | None | Switched to pitch loop #2 |
| 3/15/71 Orbit 1182 | MWA motor #1 brush wear rate excessive (16 mils/ 1 rev) | | | |
| 3/15/71 Orbit 1182 | MWA motor #1 drive voltage increased from 14 V to 22.5 V | | | |
| 3/15/71 Orbit 1182 | MWA temperature increased from +27°C to over +50°C | Brush debris shorting motor commutator | Momentary losses of lock | None at this time |
| 3/15/71 Orbit 1182 | MWA motor #2 brush wear rate excessive (6.5 mils/ 6 revs) | | | |
| 6/16/71 Orbit 2340 | MWA temperature began upward trend from +29°C | | | |
| 7/16/71 Orbit 2714 | MWA temperature peaked above +50°C | | | |
| 1/5/71 Orbit 317 | ITR[b] dropped out clock signal; several events of this type occurred leading to: | Magnetic tape sticking to magnetic head | - - | - - |
| 5/28/71 | Failure of ITR | Magnetic tape sticking to magnetic head | Loss of data from experiments | None; this occurred after required 6-month mission life |

[a]MWA - Momentum Wheel Assembly

[b]ITR - Incremental Tape Recorder

Table C. 30-2. On Orbit Malfunction Summary - ITOS A (NOAA 1), Launched 12/11/70 (Cont.)

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 1/6/71 Orbit 312 | SR[c] #1 failed; this was 5-1/2 months after expected on-orbit life | Mechanical failure within drive mechanism of mirror scan | Loss of SR redundancy only | Change of lubrication processing by the vendor and relubrication of bearing for later SRs |
| 3/16/71 Orbit 1182 | Pitch loop #1 failed (4 months after required mission life) | Collection of brush debris on motor #1 commutator | Pitch loop redundancy lost only | MWA changed to a brushless motor configuration on ITOS B and later spacecrafts |
| 7/16/71 Orbit 2720 | Pitch loop #2 failed (8 months after required mission life) | Collection of brush debris on motor #2 commutator | Pitch lock lost (end of mission) | |

[c]SR - Scanning Radiometer

## Table C. 30-3. On-Orbit Malfunction Summary - ITOS D (NOAA 2), Launched 10/15/72

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 10/15/72 | SR[a] recorder 3 transport pressure observed to continuously decrease in orbit | Leak in pressurized transport container seal | None; SR recorder 3 continued to operate normally throughout mission life of S/C | None; possibility of pressure leak recognized prior to launch; adequate post-launch recorder life projected from estimated leak rate |
| 10/15/72 | S/C nutation damping performance indicated higher than predicted time constant for liquid dampers | Probably due to a destabilizing influence excited by dynamics of MWA[b] bearing free motion | None; adequate nutation damping has been obtained in orbit | Investigations into this matter have been conducted; MWA bearing play measurements are now performed on all MWA systems |
| 10/15/72 | On-orbit attitude drift tests indicated a considerably larger S/C residual dipole than measured prior to launch | Not known; may be caused by (1) error in pre-launch measurement technique or (2) magnetization of S/C after measurement | None; compensation of the residual dipole accomplished via mag. bias switching and unipolar QOMAC[c] programs | Improvements have been incorporated in mag. dipole measurement techniques |
| 10/24/72 | Turn-on of either S-band transmitter or the VHRR[d] recorder while SR recorder 3 is recording may momentarily disturb recorded data | Probably a transient disturbance of 2400 Hz SR recorder 3 clock input line causing recorder speed change | Brief loss of digital data at time of turn-on of offending units; this represents an insignificant loss over an orbit's time | None; problem appeared to be unique to ITOS D |
| 10/26/72 | On two occasions, pitch loop #1 motor current and voltage observed to decrease to near zero for one sample period each | Not known; possibly false recognition of horizon pulse by pitch loop | None; event did not disturb S/C pitch attitude | None |
| 10/30/72 | The VHRR 1 & 2 outer frame temperatures ran warmer by 16 to 25°C than originally predicted | Disparities in heat factors used in thermal prediction computer model | None; critical inner frame temperatures remained within desired values | Heat input factors refined and incorporated into thermal prediction computer model to provide good prediction and flight temperature agreement |

[a]SR - Scanning Radiometer
[b]MWA - Momentum Wheel Assembly
[c]QOMAC - Quarter Orbit Magnetic Attitude Control
[d]VHRR - Very High Resolution Radiometer

Table C. 30-3. On-Orbit Malfunction Summary - ITOS D (NOAA 2), Launched 10/15/72 (Cont.)

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 11/02/72 | Telemetry from S/C indicated S-band transmitter 2 operated 10°C hotter than S-band transmitter 1 | Telemetry calibration curve used for S-band 2 erroneous | None | Correct calibration curve replaced original one |
| 11/03/72 | VREC[e] motor current increased from 400 mA to 680 mA coincident with change in selection of TBU[f] | Switch of TBU while VREC is recording will cause a change of phase between rotor and stator of recorder hysteresis synchronous motor; demagnetization of rotor also possible | None | Operational personnel cautioned against switching TBUs while VREC operating |
| 11/22/72 | VHRR motor current telemetry sensitive to variations in S/C unregulated load bus voltage | Parasitic oscillation in -21 V VHRR power supply regulator | None; only effect was to cause inaccuracies in telemetered motor current; VHRR data not affected | Design modification made to VHRR to suppress oscillations |
| 01/06/73 | An SR recorder playback over the Gilmore CDA terminated prematurely before recovery of all data on tape | Cross-coupling of relay signals within CDU[g] allowed S-band playback on command transmitted after transmitter had already been turned on, to terminate the playback mode | One-time loss of part of data on tape; no effect on overall mission | Retransmission of the subject command after the transmitter is already on is not normal command procedure; operational personnel were cautioned, and reoccurence of this event is not expected |
| 03/19/73 | Jitter observed in VHRR, visible pictures | Jitter caused by uncertainty in reclocking the once-around pulse when pulse is near clock edge; pulse location drifts with age | VHRR imagery slightly degraded by 200 μsec jitter; problem disappears when once-around pulse drifts out of coincidence with clock edge, usually a few days | Redesign of once-around decoder circuits is under consideration; problem not considered serious since it appears to last for short interual and is non-recurrent during normal life of instrument |

[e]VREC - Visual Data Recorder
[f]TBU - Time Base Unit
[g]CDU - Command Data Unit

Table C. 30-3. On-Orbit Malfunction Summary - ITOS D (NOAA 2), Launched 10/15/72 (Cont.)

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 04/11/73<br>04/19/73<br>05/11/73 | (1) S/C failed to execute a command; receipt of command by decoder was verified<br><br>(2) S/C accepted an address retransmission as a valid command<br><br>(3) S/C responded erroneously to a command sequence | Occasional command problems experienced by S/C caused by unfavorable atmospheric conditions during transmission resulting in pulsating signal strength at S/C; this causes dropouts in detected data ones and zeros resulting in decoder misinterpretation of commands | This occasional occurrence may result in wrong operating modes on S/C until corrected by follow-on commands; therefore, the effect is only temporary; there is no possibility of a misinterpreted commend causing effects from which the S/C cannot recover; main impact is of a nuisance type | None for immediate use; later S/C of the H series will use decorder of modified design that will be less susceptible to these command anomalies |
| 01/03/74 | VTPR[h] 1 failed to operate; started operating again on 6/13/74 and continued to provide good data until it failed again on 11/11/74; initial failure occurred during 15th month in orbit, or 9 months after required mission life | Believed caused by filter motor bearing lubrication problem or binding in gear drive | VTPR 2 continued to provide mission data until it failed 3 months later; at that time NOAA 3, which was already in orbit, was designated the operational S/C and provided the VTPR mission data | Gear box lubricant was changed from P-10 oil to Krytox in later VTPR's; in still later change, filter-chopper drive system was completely redesigned |
| 03/18/74 | VTPR 2 failed to operate; failure remained until deactivation of S/C; failure occurred during 17th month in orbit, or 11 months after required mission life. | Believed caused by filter motor bearing lubrication problem or binding in gear drive | See above | See above |
| 03/12/74 | VHRR 2 started to show periodic loss of motor speed lock-up; frequency of occurrence increased with time time; eventually it would not lock up at any time | Unexplained; considered the result of 17 months of in-flight aging | In May, VHRR 2 was turned off and only VHRR 1 was operated in a reduced back-up mode; the NOAA 3 S/C became prime for VHRR data | None; VHRR 2 had been operated continuously since turn-over to NOAA after launch, representing 17 months of flight life, or 11 months after required mission life |

[h]VTPR - Vertical Temperature Profile Radiometer

Table C. 30-3. On-Orbit Malfunction Summary - ITOS D (NOAA 2), Launched 10/15/72 (Cont.)

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 04/74 | VHRR 1 started showing signs of periodic loss of motor speed lock-up; condition observed 12 months after required mission life | Unexplained; probably due to old age | NOAA 3 S/C made prime for VHRR data; VHRR 1 on NOAA 2 continued to operate with only short periods of disturbances until deactivation of S/C in 1/75 | None; VHRR 1 had been operated continuously since turn-over to NOAA for over 18 months of flight life before first signs of motor speed instability developed; data prom unit remained usable until S/C deactivation |
| 10/28/74 | Battery 2 temperature started a gradual increasing trend; condition appeared after 24 months in orbit, or 18 months after required mission life | Unexplained; believed due to change in battery 2 characteristics as consequence of aging, or a partial cell short | This event contributed to final deactivation of S/C | Battery commanded into trickle charge to limit charge current and thereby maintain normal temperatures; in this condition, the S/C load support capability is limited |
| 01/30/75 | S/C deactived after 2.3 years in orbit | Failures of VHRR 1, VHRR 2 and high temperature of battery 2 | NOAA 3 and NOAA 4 S/C continued to provide mission requirements | |

Table C. 30-4. On-Orbit Malfunction Summary - ITOS F (NOAA 3), Launched 11/6/73

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 11/24/73 | Position code data read out by VTPR[a] 1 to identify associated radiance words became errotic; did not disturb radiance data | Not known; believed associated with position code counter logic which appeared to be counting erratically by twos | VTPR 1 still able to supply mission data; condition occurred 5 months earlier than required mission life | User program modified to eliminate dependence on position code identifiers |
| 11/26/73 | VTPR 1 hung up in patch-look calibration position; instrument was turned off to allow return to scan; later, on 2/22/74, VTPR 1 would not enter electrical calibration sequence; calibration function has remained inoperative since | Not known; believed to be electrical and in the calibration sequencer portion of instrument; possibly associated with 16 PPS .2 PPS signal flow | VTPR 1 still able to supply mission data; condition occurred 5 months earlier than required mission life | User established calibration from previous calibration sequences updated with pitch-up of S/C to give space look calibration references |
| 03/03/74 | SR[b] 2 motor rotation stopped; failure occurred during 4th month in orbit, and 2 months before required mission life | Depletion of lubricant in motor bearings | None; second redundant SR was still operational | Used SR 1 for mission |
| 06/02/74 | VTPR 2 filter wheel rotation stopped; failure occurred 1 month after required mission life | Bearing or gear box lubrication problem | None; VTPR 1 used to perform mission | For later VTPRs, changed gear box lube; still later, redesigned filter-chopper drive system |
| 08/07/74 | 200-μsec jitter appeared in timing and calibration signals in each VHRR[c] 1 line scan; after 2 days, jitter problem disappeared | Problem caused by reclocking uncertainty when once-around line sync pulse and clocking pulses are near coincidence; reclocked once-around initiates counter that establishes location of timing signals | VHRR imagery slightly degraded by 200-μsec jitter; problem disappears when once-around pulse drifts out of coincidence with clock edge | Redesign of encoder to improve once-around stability is under consideration; once in orbit, only corrective action is to wait for once-around to drift away from clock edge, usually a few days |

[a]VTPR - Vertical Temperature Profile Radiometer

[b]SR - Scanning Radiometer

[c]VHRR - Very High Resolution Radiometer

Table C. 30-4. On-Orbit Malfunction Summary - ITOS F (NOAA 3), Launched 11/6/73 (Cont.)

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 09/14/74 | Same type of problem reported for VHRR 1 above appeared in VHRR 2; after 1 week, jitter problem disappeared and has not been reported since for either radiometer | Problem caused by reclocking uncertainty when once-around line sync pulse and clocking pulses are near coincidence; reclocked once-around initiates counter that establishes location of timing signals | VHRR imagery slightly degraded by 200-μsec jitter; problem disappears when once-around pulse drifts out of coincidence with clock edge | Redesign of encoder to improve once-around stability is under consideration; once in orbit, only corrective action is to wait for once-around to drift away from clock edge, usually a few days |
| 04/05/74 | As consequence of pitch-up maneuver performed on this date, existence of a radiometric bias was discovered in VTPR data while viewing space | VTPR field of view was broader than previously defined, allowing portions of S/C structure to contaminate accepted input radiance | Contributed to larger than expected errors in temperature sounding data | (1) S/C pitch-up procedure to view space and thereby determine values of bias correction factors on an individual instrument basis made part of in-flight operations<br>(2) Design mod: optical baffle added to VTPR primary optics to correct field of view |
| 01/10/74 to 03/08/74 | Both VTPRs and SRs observed to operate at hotter than predicted temperatures especially during the 100% sun-time period of orbit | Errors in analytic thermal model predicted lower temperatures than achieved in flight | None; however temperatures were within a few degrees of point where radiometer performance could degrade | Thermal modifications incorporated in later S/C to lower operating temperatures by 3-5° and therefore pickup more margin; modifications included blanket cutouts, change of paint, and addition of radiators |
| - - | Streaking in imagery produced from VHRR 2 IR video caused by line-to-line intensity variations; condition present from beginning of life, but is only visible when IR-2 is selected | Under investigation; not known at this time | User switched to VHRR 1 IR video, which had no perceptible streaking | To be determined from proposed investigative program |

Table C. 30-4. On-Orbit Malfunction Summary - ITOS F (NOAA 3), Launched 11/6/73 (Cont.)

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 05/13/74 | A rephasing word command was improperly interpreted by S/C as a two word command | Probably noise or RF fading during command transmission caused improper shifting of command bits into decoder message register | Slight shift in location of VREC[d] coverage for one orbit; no long term effects | During the redesign of decoder to use TTL logic for ITOS H, I, and J contract, certain logic changes were incorporated to reduce the probability for this and other known command anomalies to occur |

[d]VREC - Visual Data Recorder

Table C. 30-5. On-Orbit Malfunction Summary - ITOS G (NOAA 4), Launched 11/15/74

| Date | Malfunction | Cause | Impact on Mission | Corrective Action |
|---|---|---|---|---|
| 12/25/74 | Loss of SPM[a] channel 10 (SPM is GFE) | Not known | Remaining 5 sensors continue to furnish operational data; failure occurred before second month on orbit; loss of the SPM secondary sensor data in one channel | None |
| 02/04/75 | Evidence of some noise in SR[b] data recorded on SR recorder 3; it appears as occasional short white bursts in picture products | Noise is actually in F&W channel; it is coupled into SR video data by ground performed Z axis correction | NOAA considers SR data recorded by SR recorder 3 to be usable and causing no difficulties in processing; not a mission problem | None |

[a] SPM - Solar Proton Monitor

[b] SR - Scanning Radiometer

## C.31 EARTH RESOURCES TECHNOLOGY SATELLITE (ERTS/LANDSAT)

### C.31.1 Program Summary

The ERTS program (renamed LANDSAT in 1975) was established by NASA for the purposes of:

a. Determining those earth resources data that can be acquired best from spacecraft.

b. Determining how the increased frequency, synoptic, and global coverage uniquely afforded by spacecraft observations can aid the study of time-variant and relatively unchanging phenomena.

c. Determining which earth resource data can be most effectively and economically obtained by (1) manned, unmanned, or manned-serviced satellites; (2) relay of surface sensor data via space; or (3) current means now being used.

d. Developing improved methods of displaying (mapping) and disseminating space-acquired earth resource data on a global basis suitable for utilization by scientific and commercial interests.

e. Discovering what unforeseen earth resources or geoscience phenomena may be observable from the overview available at orbital altitudes.

The ERTS contract called for two experimental satellites, with the Space Division of General Electric Company as the major contractor. The first ERTS satellite was launched in July 1972 and the second in January 1975, each on a Delta booster into near-earth polar orbits.

Specific requirements of the ERTS system are to provide:

a. Multispectral photographic and digital data of a large scene in user-oriented quantities

b. Repetitive land and coastal observations at the same local time

c. Image overlap in direction of flight

d. Image sidelap from adjacent orbits

e. Image location to better than 2 nmi

f. World coverage in less than 3 weeks

g. Operating spacecraft life of 1 year

h. Facilities for processing and distributing data to investigators in useful form and on a timely basis

### C.31.2 Satellite Description

The ERTS program utilizes the spacecraft originally developed for the Nimbus program. The configuration and general features of the satellite are shown in Figure C.31-1, with supplemental information in Tables C.31-1 and C.31-2. Specific details of subsystems, including some subsystems peculiar to the ERTS spacecraft, are given in the discussion of the Nimbus satellites (Section C.28).

### C.31.3 Key Events and Milestones

Significant milestones in the ERTS program, including on-orbit problems, are summarized in Figure C.31-2.


LAUNCH DATES
ERTS A - 23 JULY 1972
ERTS B - 22 JAN 1975

BOOSTER - DELTA 900
ORBIT - 500 NM
$99^{\circ}$ INCLINATION

CHARACTERISTICS

| | | |
|---|---|---|
| WIDTH (ASCENT) | - | 60 IN |
| WIDTH (DEPLOYED) | - | 156 IN |
| HT (OVERALL) | - | 120 IN |
| WT (LIFTOFF) | - | 2073 LB |
| POWER (BOL) | - | 550 WATTS |
| DESIGN LIFE | - | 1 YEAR |

Figure C.31-1. Earth Resources Technology Satellite (ERTS/LANDSAT)

Table C.31-1. ERTS/LANDSAT Technical Details

**Satellite**

- 10 ft high × 5 ft diameter
- 13 ft across array
- 2100 lb

**Attitude Control**

- 0.7° in three axes
- 0.015° /sec rate

**Power**

- Solar array - 550 W
- Batteries (8) - 36 A-hr each

**Tracking, Telemetry, and Command**

- MSFN and STDN compatible
- 912 TLM points
- 520 commands

**Thermal Control**

- Passive and active (shutters)
- 20° ± 10°C

Table C.31-1. ERTS/LANDSAT Technical Details (Cont.)

Attitude Measurement Sensor

- $0.07^{\circ}$

Wideband Video Tape Recorder (WBVTR)

- 30 min capacity - RBV and MSS data

Wideband Telemetry

- 20-W S-band - RBV, MSS, and WBVTR

Orbit Adjust

- 260 ft/sec - in-plane and inclination errors

Launch Data

- Launches
  - ERTS A - 23 July 1972 (LANDSAT I)
  - ERTS B - 22 January 1975 (LANDSAT II)
- Delta launch vehicle
- Polar, sun-synchronous orbit

Developed by

NASA

General Electric

Table C.31-2. ERTS/LANDSAT Payload

Return Beam Vidicons (RBV)

- Resolution - 100 meters (per line pair)
- Coverage - 100 × 100 nmi (185 km)
- Spectral bands - 0.475–0.575, 0.58–0.68, 0.69–0.83 μm
- Data - video, 3.5 MHz

Multispectral Scanner (MSS)

- Resolution - 160 meters (per line pair)
- Coverage - 100 nmi swath (185 km)
- Spectral bands - 0.5–0.6, 0.6–0.7, 0.7–0.8, 0.8–1.1 μm and 10.4–12.6 μm (ERTS B)
- Data - PCM 15 Mbps

Data Collection System (DCS)

- Ground platform input - 8 analog signals
- Potentially 1000 platforms
- Data - 100 kHz

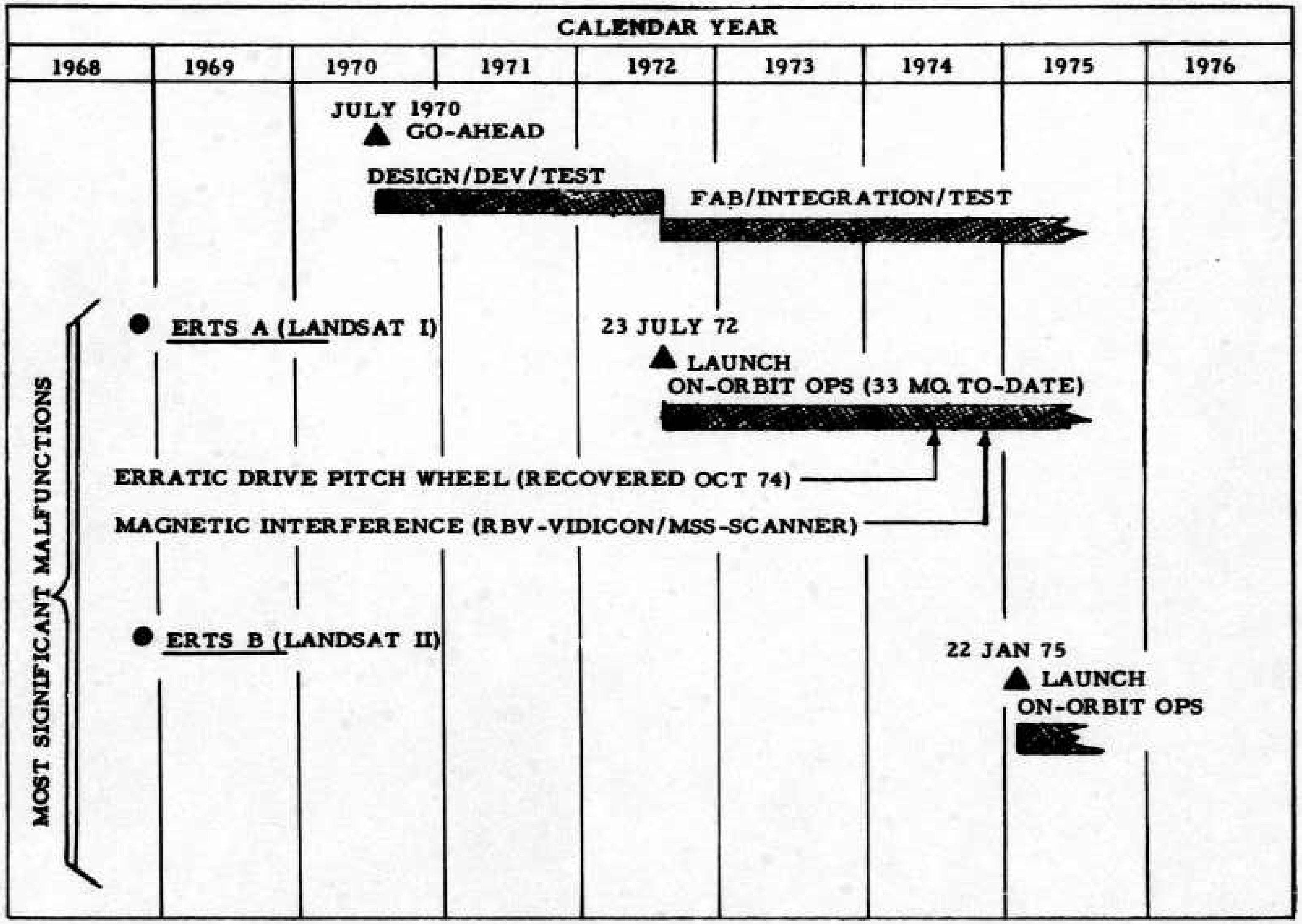



Figure C.31-2. Key Milestones and Events - ERTS/LANDSAT

AEROSPACE CORPORATION

**INTERNAL DISTRIBUTION LIST**

(REFERENCE: COMPANY PRACTICE 7-21-1)

| REPORT TITLE | | |
|---|---|---|
| A Review of Communications Satellites and Related Spacecraft for Factors Influencing Mission Success - Volume II: Appendices A, B, C | | |
| REPORT NO. | PUBLICATION DATE | SECURITY CLASSIFICATION |
| TOR-0076(6792)-1, Volume II | 17 November 1975 | Unclassified |

(NOTE: FOR OFF-SITE PERSONNEL, SHOW LOCATION SYMBOL, e.g. JOHN Q. PUBLIC/VAFB)

R. E. Austin
G. W. Bailey
A. D. Bredon
F. W. Buehl
M. A. Clark
J. R. Coge
R. A. Davis
E. H. Edwards
J. J. Egan
F. R. Gerardi
R. E. Hammerand
T. H. Hanrahan/CB
D. O. Heitkamp
F. L. Hennessey
A. D. Halenbeck
V. Josephson
C. H. Kelley
C. M. Kelly
G. M. Kelsey
W. J. Kleinbauer
W. A. Knittle
W. F. Leverton/J. Statsinger
S. E. Levine
D. H. Martin
F. S. Nishime
A. C. Reed
E. I. Roberts
D. J. Scrooc
L. R. Sitney
J. D. Sorrels
V. W. Wall/H. E. McDonnell
J. R. Wood/VAFB
C. J. Zamites

APPROVED BY Samuel E. Levine DATE 11-7-75

SHEET ______ OF ______

AEROSPACE FORM 2394 REV 7-66

THE AEROSPACE CORPORATION

**EXTERNAL DISTRIBUTION LIST**

(REFERENCE: COMPANY PRACTICE 7-21-1)

REPORT TITLE
A Review of Communications Satellites and Related Spacecraft for Factors Influencing Mission Success - Volume II: Appendices A, B, C

| REPORT NO. | PUBLICATION DATE | SECURITY CLASSIFICATION |
|---|---|---|
| TOR-0076(6792)-1, Volume II | 17 November 1975 | Unclassified |

MILITARY AND GOVERNMENT OFFICES — ASSOCIATE CONTRACTORS AND OTHERS

(NOTE: SHOW FULL MAILING ADDRESS; INCLUDE ZIP CODE, MILITARY OFFICE SYMBOL, AND "ATTENTION" LINE.)

SAMSO
A.F. Unit Post Office
Los Angeles, California 90045

D. W. Dodd (AWL)
Col. R. D. Greiner (AW) (2)
Col. R. W. Johnson (DY)
Col. B. J. Meyer (SK)
Col. F. S. McCartney (SKA)
Col. G. J. Murphy (LV)
D. J. Newman (AWZ)
Col. K. O. Sameshima (IN)
L/C J. C. Saylor (LVC)
Col. W. L. Schlosser (XR)
Col. C. Zimmerman (DYE)

Navy Space Systems Activity (NSSA)
A.F. Unit Post Office
Los Angeles, California 90045

Cdr C. S. Bond (NSSA) - 4 copies

Department of Defense
Defense Communications Agency (DCA)
Code 800
Washington, D. C. 20305

Cdr W. H. Curry, Jr. - 100 copies

[X] THIS DISTRIBUTION LIST PERMITS ONLY PRIMARY DISTRIBUTION OF THIS TOR.

AFR 80-45 DISTRIBUTION STATEMENT X'D BELOW APPLIES

[ ] NO DISTRIBUTION STATEMENT (Classified documents only)

[ ] A. APPROVED FOR PUBLIC RELEASE; DISTRIBUTION UNLIMITED

[ ] B. DISTRIBUTION LIMITED TO U. S. GOV'T AGENCIES ONLY; ____ (Reason); ____ (Date statement applied). OTHER REQUESTS FOR THIS DOCUMENT MUST BE REFERRED TO ____ (Controlling DOD office).

APPROVED BY Samuel E. Lamm (FOR THE AEROSPACE CORPORATION) DATE 11-7-75

APPROVED BY Bobby J. Meyer (FOR COGNIZANT AF OFFICE) SK (SYMBOL) DATE 10 Nov 75

IF LIST COMPRISES TWO OR MORE SHEETS, COMPLETE THIS SIGNATURE BLOCK ON LAST SHEET ONLY

AEROSPACE FORM 2380 REV 7-71

SHEET ____ OF ____